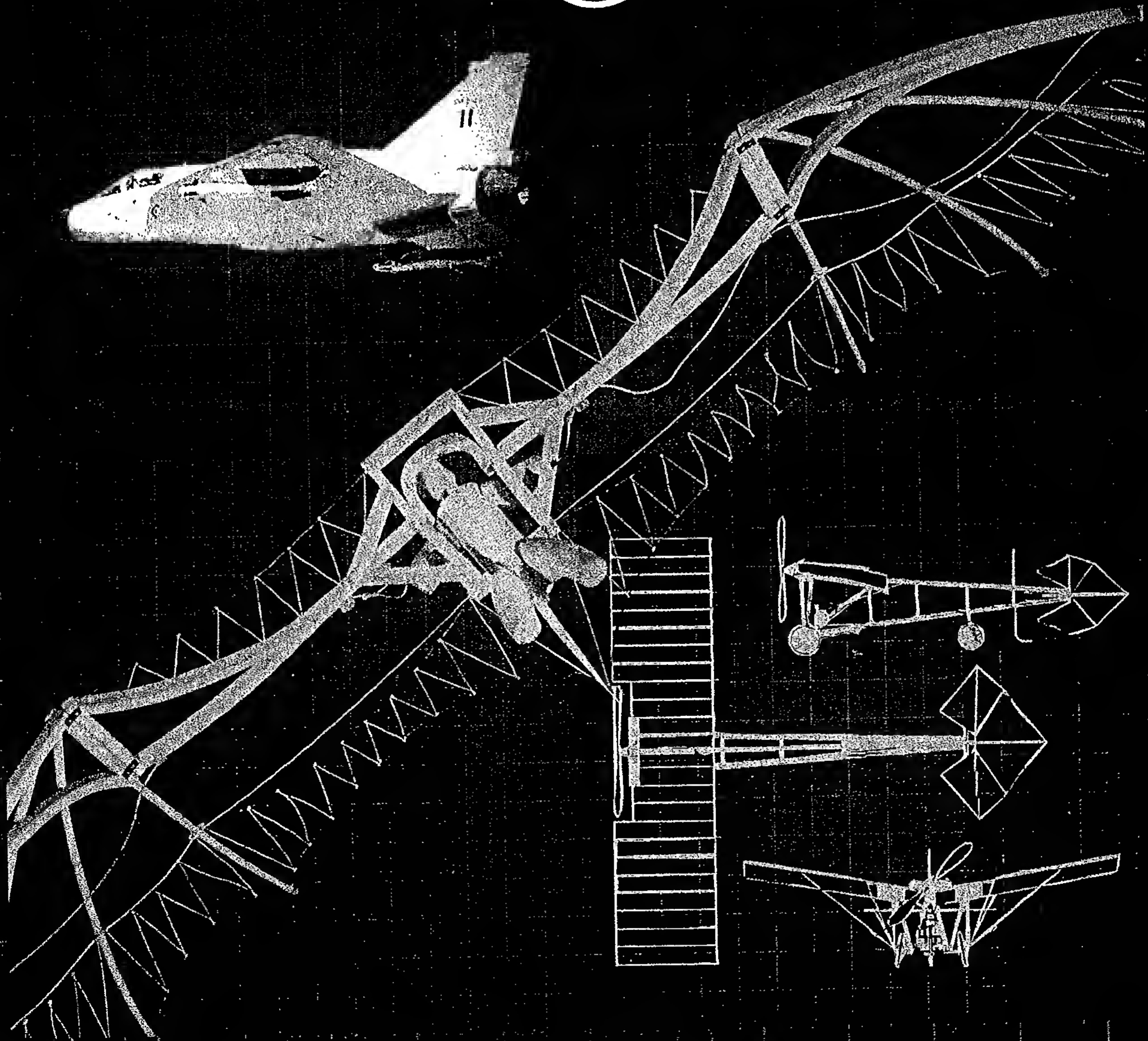
ARMS 2008

ABSTRACTS OF THE PROCEEDINGS OF

Sixth National Seminar And Exhibition On Aerospace And Related Mechanisms

28 - 29 March 2008



ORGANISED BY

Indian National Society For Aerospace And Related Mechanisms &
Armament Research And Development Establishment, Pashan, Pune - 411 021

डीआरडीओ स्वर्ण जयंती वर्ष

DRDO Golden Jubilee Year

रक्षा अनुसंधान तथा विकास संगठन

बलस्य मूलं विज्ञानम्

रक्षा मंत्रालय

1958 - 2008

सेवा में समर्पित 50 वर्ष

50 Years Of Dedicated Service To The Nation

FOREWORD

Aerospace mechanisms are complex and mission critical, requiring high precision, accuracy and reliability, ranging in applications from sensors to actuators to complex space launch systems. The applications of mechanisms in satellites, missiles and weapon systems are challenging in view of the need for perfection in design and development, predictability and optimization.

ARDE, Pune is organizing ARMS 2008, the Sixth National Seminar on Aerospace and Related Mechanisms from 28-29 March 2008 under the auspices of INSARM. DRDO and ARDE are celebrating Golden Jubilee during 2007-2008. It is befitting that the seminar is being arranged as a part of the Golden Jubilee celebrations.

It is heartening to note that there has been an overwhelming response to the seminar in the form of more than 100 technical papers in the fields of design and development of mechanisms, testing & evaluation and reliability. The interesting mix of papers indicates the emerging trends and thrust areas especially in the fields of ejection, deployment and recovery systems in spacecraft launch systems, aircraft, rockets, submunitions, etc. I am sure this seminar will serve as a platform for exchange of ideas and sharing of expertise among scientists, technologists and academia engaged in design integration and test & evaluation of mechanisms in aerospace and ground systems.

The Pune Chapter of INSARM is being inaugurated during the seminar, which will help to promote the participation by Region-based Industry, Defence Laboratories and other participants in development of critical specialized mechanisms.

The seminar will benefit all the participants through fruitful deliberations for progress in the fields of mechanisms which will lead to self-reliance in defence and space technologies.

(SURENDRA KUMAR)
DIRECTOR ARDE

1. INTRODUCTION

The Indian National Society for Aerospace and Related Mechanisms (INSARM) has been organizing seminars once in three years. The fifth seminar was conducted at ISRO Satellite Centre, Bangalore, in Dec 2005. At present there are three chapters of the society namely at Triruvananthapuram, Bangalore and Hyderabad. The fourth chapter at Pune would be inaugurated during ARMS 08 Seminar.

2. ARMS 2008

The sixth INSARM seminar is being organized at Armament Research & Development Establishment, Pune, on 28-29 Mar 2008. This seminar is jointly sponsored by DRDO and ISRO, and co-sponsored by many Government Agencies, Industries, R&D and Academic Institutions in India.

3. OBJECTIVES/AIMS OF THE SEMINAR

- To focus on recent advances in design, fabrication, testing and analysis of aerospace and related mechanisms
- To study future concepts related to aerospace and related mechanisms
- To address the challenges faced by the 'Mechanisms' community

4. TOPICS

The seminar topics will cover mechanisms for satellites, launch vehicles, missiles, aircrafts, interplanetary missions, military applications, scientific and commercial applications, with the following areas of specific interest.

- Mechanisms of launch vehicles/missiles / aircrafts / helicopters / satellites
- Automatic mechanisms in armaments, aircraft guns and munitions
- Smart mechanisms / MEMS / Mechatronics for aerospace and armament applications
- Ejection systems
- Fuzing mechanisms
- Recovery systems
- Deployment mechanisms
- Pointing mechanisms
- Tribology for aerospace and armament applications
- Design, testing, instrumentation & analysis
- Digital prototyping
- Robotics
- Safety & Reliability
- Problems and challenges in aerospace and related mechanisms

5. TECHNICAL PAPER PRESENTATION

Out of 97 papers received from more than 20 institutes of DRDO/ISRO and allied agencies, 42 papers will be presented at the seminar. The technical papers will be presented in 10 sessions, organized, under four subject categories viz. Design & analysis of mechanisms; Ejection, deployment and recovery systems; Development, testing & evaluation; and Safety & reliability. Poster presentations for 55 papers is another highlight of the seminar.

6. EXHIBITION

An exhibition of newly developed mechanisms and related products will be a special attraction at the seminar venue. Exhibits of mechanisms, working models, software related to mechanisms, laminat ed phots, products and posters will be on display. Important products developed by six DRDO/ISRO establishments will occupy pride of place at the exhibition. Also on display will be a variety of exhibits using surface engineering methods, coating technologies and other mechanisms developed by eleven private firms.

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ABSTRACTS

ABSTRACTS

Design & Analysis of Mechanisms

Challenges Faced during Development of A Disconnect Mechanism for Explosive Transfer Lines

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Explosive Transfer Assembly (ETA) lines are employed in the pyro circuit of various critical explosive systems such as stage separation systems, ignition systems, destruct systems, heat shield separation systems etc. A typical pyro circuit consists of a safe/Arm and explosive manifold from which ETA lines are branched out to various terminal devices which require simultaneous triggering. There are certain applications where a common pyro circuit has to trigger the functioning of different explosive devices/ systems mounted on adjacent stages of a launch vehicles. In such cases, the ETA lines run across the separation plane between the two stages. Disconnection of these ETA lines is a mandatory functional requirement to ensure smooth separation of spent stages in a normal flight. A disconnect mechanism for ETA lines has been developed for this purpose.

The functions of ETA disconnect mechanism are to ensure positive transfer of explosive stimulus to explosive systems/devices mounted on different stages, withstanding flight environments and to ensure smooth disconnection of upper and lower parts of ETA lines during separation of spent stage. In case of application in destruct system pyro circuit, the disconnected live ETA lines should be protected from inadvertent firing due to the hot gases from ongoing vehicle.

ETA disconnect mechanism essentially consists of two adaptors where ETA lines which are to be coupled are assembled. These adaptors mounted on two housings are mated and kept in position by means of a helical compression spring designed to meet the flight vibration levels. The two housings are mounted on the upper and lower stages respectively and are mated across the separation plane. Adaptors along with their housings are de-mated during stage separation.

During the development and qualification of the mechanism, tests were conducted at various off-nominal conditions to demonstrate satisfactory performance meeting all the intended functional requirements. Interface gap between the ETA lines, axial misalignment between the ETA adaptors, taper angle for smooth demating of ETA adaptors, thermal simulation to ensure safety of disconnected ETA lines etc were some of them. The mechanism was successfully developed, qualified and used in ISRO launch vehicles.

This paper deals with the description of the mechanism and the various off-nominal tests carried out to meet the challenges faced during development and qualification of the mechanism.

Design and Development of a Pyrothruster for Deep Space Application

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Pyrothruster is basically a Cartridge Actuated Device (Cad) meant for transferring thrust over specified distance, used mainly in separation systems. ISRO also uses Pyrothrusters in conjunction with a Ball Lock separation system in its launch vehicles. It is essentially a piston-cylinder device powered by a pyrotechnic charge having the advantages of low mass, small size, low power requirement etc making it the abovious choice for similar applications. For one of iSRO's future missions a new Pyrothruster is required for operation in deep space. The location of the Pyrothruster in the vicinity of highly sensitive equipments mandates complete hermetic sealing of the pyro combustion products within the device and prevention of outgassing from any such constituents of the device. Existing Pyrothrusters were not designed for deep space application and rely only on dynamic O'rings to prevent seepage of combustion products across the piston. Moreover these O'rings cannot ensure complete containment of combustion products and are prone to outgassing and mass loss in deep vacuum; the existing desgings could not be used as such in deep space necessitating extensive reconfiguration/redesign. This task was made further difficult with stringent envelope constraints. This paper deals with the challenges involved in achieving all the new requirements with minimum modifications of the existing design, which is having considerable design heritage, and still meeting the interface and envelope constraints of the system.

DM/O/02

Problem of Thin Section Ball Bearing at Cold Temperature for a Typical Space Mechanism.

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Thin section ball bearings are used in Space Mechanism where loads are low and also when light weight bearings are required. Because of their thin cross section they are sensitive to loads from the housing and can flex leading to jamming of the bearings. In space application when these bearings are used in system where they are located on outer surface of satellite, the bearings are exposed to cold temperature. Under these conditions the housing of the bearings also plays an important role. If housings contract they load the bearings causing radial flexing and consequently reducing the clearances in the bearings. Under this condition the bearings are not able to freely rotate. Functioning of the system is affected.

The paper describes, the temperature experienced in the orbit, the finite element analysis of the bearings, the tests carried out, the mission management, the choice of materials and the solution to such problems.

Temperature is calculated based on the space environment. The finite element solution includes the simulation of balls in the bearings and the local deflection has been calculated. The tests have been carried out in a hot and cold chamber simulating the bearing and its housing. The temperature of the housing can be increased by exposing towards sun by reorienting the spacecraft. The choice of materials can be by use of Titanium as the bearing is made of steel. All these aspects have been discussed in the paper. The paper also describes the hardware details. This can be a lesson to a designer.

DM/O/03

Design of Releasable Support Pad Assembly for Canister Launched Large Size Flight Vehicle

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In canister launched missile systems, flight vehicle is supported by specially designed pads. They exist in the gap between canister and missile during loading, storage and launch operations. These pads are meant to mitigate the bearing pressure on the missile. Shortly after the missile leaves the canister, due to the pressure produced by gas generator, the role of pads ceases to exist. Hence it is desirable to get these pads jettisoned as the missile leaves canister. Enough information to corroborate availability of variety of missile support pads, design aspects of them are not well addressed. This note attempts an innovative design or releasable pads for a canisterised version of a large sized flight object.

There are two aspects in designing any release mechanism for canister system. First aspect governs design, energy storage for jettisoning, and restraining them at the desired location on the missile. This gives configuration of pads, and release mechanism principle. The other aspect covers selection of proper material to ensure storage and loading of flight object into canister. Further, it is aimed for an automatic, disturbance free disengagement of the support pad arrangement from the flight vehicle. A design methodology satisfying these requirements Is evolved after design trade off studies. This paper deliberates on issues and challenges in designing a release mechanism and its assembly with the launch vehicle.

Design of pad is critical, as the safe stay of flight object inside canister is inextricably linked with pads robustness. Constraints like accommodating pads in the gap between flight object and canister, ensuring self alignment, strength to have acceptable bearing pressure on the flight object as well as canister with minimum deflection of the complete assembly are discussed. Outer surface of the pads interact with the inner surface of the canister which is primarily a thermal protection composite layer. Distribution of the predicted reaction force on this layer viz a viz friction factor, material selection, related tribological and corrosion issues are presented. The key energy storage device in this design is spring steel band. Load bearing pads are attached on this band. This combination embraces the circumference of the cylindrical portion of the missile after final assembly. By assembling the band on the missile in a particular fashion, required energy is stored and retained. Issues in deciding the length of the band and energy storage method is also addressed.

This paper outlines design of a typical release mechanism for canister launch system, covering major challenges like restraining pad assembly on the flight object during loading into (or out of) canister, low friction factor, its integration, automatic sensing of release time, and releasing without getting entangled with other bands.

DM/O/04

ESD Aspects of CFRP Honeycomb Sandwich Payload Fairing

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Composites have been in space application almost from the start of the space program. They have been used in aircrafts, launch vehicles, missiles and spacecraft structures. High performance composites satisfies the requirements of space structures like high stiffness, low coefficient of thermal expansion and dimensional stability. But their sensitivity to temperature, moisture and space environment is different from that of metal counterpart. Payload fairing is a critical aerodynamic structure which accommodates the payload / space-craft and protect it from temperature, humidity, dust and atmospheric fields while on pad and from thermal, acoustic, aerodynamic and tribo charging while on launch and ascent. One of the launch vehicle structure for which composite replacement, against traditional aluminium alloy, tried out by most of the countries for better payload mass fraction (structures with reduced structural weight will increase the payload mass fraction) and low cost ELV (Expendable Launch Vehicle), is payload fairings. USA, Europe, Russia, Japan are few countries that had already successfully flown composite payload fairings.

ISRO is now planning to have its fully composite payload fairings for its GSLV-D3 mission and for its future LV-mkIII missions. Over the advantages of composites payload fairings, extremely rigid and essential characteristics for protecting satellite/ payloads, the main issues include lower conductivity, handling damage and variability in raw material. Lower conductivity is a major concern of composite structure resulting in poor electrical grounding, shielding and charge accumulation which may result in catastrophic failures. Their electromagnetic shielding capability is also much less than aluminium, This increases susceptibility and vulnerability of avionic equipment located inside the structure. On pad and during ascent phase there exists number of ways the vehicle gets charged as high as 10k volts. Hence it is mandatory to study the electrical conductivity characteristics of CFRP (Carbon Fiber Reinforced Polymer) and to provide control measures for static charge accumulation and discharge. In this paper review on launch vehicle static charging phenomenon, control measures for charge accumulation and static charge discharge, few test results of composite material used for the payload fairing, material testing, evaluation and acceptance criteria for ESD (Electro Static Discharge) safe missions are presented.

DM/O/05

Evaluation of mechanisms using Axiomatic Design

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Spacecraft uses many deployable mechanisms which are mission critical appendages. These mechanisms help keep the appendages stowed (folded) during launch in order to meet launch vehicle envelope constraints and are deployed once the spacecraft reaches its orbit. As these mechanisms are mission critical, successful deployment of these mechanisms is a must for a spacecraft to achieve it mission goal. It is important that these mechanisms are robust since it has to survive in harsh environment with no reparability. To meet the above requirement sound design and design evaluation methods are used.

Axiomatic design attempts to organize design knowledge by generalizing, codifying and systematizing. Basic postulate of this approach is that there are fundamental axioms that govern the design process. Two axioms were identified by examining the common elements that are always present in good designs.

Axiom1:

The independence axiom: Maintain the independence of the functional requirements. It states that when there are two or more functional requirements, the design solution must be such that each one of the functional requirement can be satisfied without affecting the other one.

Axiom2:

The information axiom: Minimize the information content of the design. Where information content I_i for a given functional requirement is defined in terms of probability P_i of satisfying functional requirement (I_i = -log Pi) .The information axiom provides a quantitative measure of the merits of a given design among the acceptable designs in terms of independence axiom.

This paper presents a case study of design evaluation of spacecraft mechanism using axiomatic design.

DM/O/06

Scan Mirror Mechanisms for Meteorological Payloads of Insat Series Satellites

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The Scan Mirror Mechanisms form a vital link between the earth's environment and optics of most of the earth observation systems from geostationary orbit, to provide imagery of the earth's environment for weather prediction, resources management etc. Currently two types of Radiometers namely Very High Resolution Radiometer [VHRR] and CCD camera are operational onboard Indian Satellites that require distinct scan modes for mapping the earth's environment. New generation instruments namely Imager and sounder are being developed for better mapping and sounding the atmosphere incorporating twelve inch optics. The operational requirements of these mechanisms demand high linearity, repeatability and reliability. This paper discusses the dual gimbal Scan Mirror Assemblies (SMA) of Imager and sounder Payloads for use onboard the INSAT class satellites. The sounder incorporates a filter wheel mechanism and works in tandem with the Scan Mechanism. This electrical synchronization between mechanisms, complex operational modes and linearity are stringent and challenging. The challenges in mechanical design include optimal design to stagger the structural modes to avoid interaction with other systems in the satellite like momentum wheel, earth sensor etc. Filter wheel design demands a thermal gradient of 100 K between the filters and the bearings/motor while maintaining the stiffness requirements.

The methods adopted to achieve the linearity, repeatability and reliability are described. The SMAs and Filter wheel drive have been developed and qualified at IISU to meet the specifications for use meteorological spacecrafts.

Design of Tailfin Locking and Its Fin Flip Out Mechanism for an Aircraft External Store

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In a tactical environment, different methods are employed to identify and destroy the enemy targets. One of the popular and effective methods is the carriage and deployment of a "smart" store using an Aircraft. In one such mission requirement, the store is carried externally either under the wings or fuselage of the Aircraft.

One of the main modules of this store is the tail unit, which houses the tail fin and other related mechanisms. A large surface area of the tail fin is required for better considerations a larger tail fin cannot be provided. The outer dimensions of the tail fin are decided mainly by ground clearances when mounted on the Aircraft. Another important consideration is the clearance between the two adjacent stores carried along side. This necessitates that the fins should be folded during carriage mode and "flipped out" after its release from the Aircraft. This paper describes the design and development of following two related mechanisms.

- Tail fin locking mechanism
- Tail fin flip-out mechanism

Tail fin locking mechanism is required to keep the tail fin in the closed position during the carriage mode. The top link is connected by a lanyard to the pylon and gets pulled at the time of release of the store. The operation of this mechanism gives way for the fin flip-out mechanism to operate. The mechanism is designed in such a way that it gets locked in both starting and end positions.

Tail fin flip-out mechanism is required to open the tail fin, after its release from the aircraft, using a lanyard. During the carriage mode, the spring is in compressed position. This provides the required energy for flipping out the tail fin after its release from the Aircraft. Dampers have been used to get the velocity reduced at the end of the stroke. In this mechanism, the linear motion is converted to rotary motion.

The mechanism has been analyzed based on the spring-mass-damper system. Design and synthesis of this mechanism is based on a modified slider crank mechanism. Kinematic analysis of the mechanism is carried out to get the motion and position of the different link at intermediate interval of time. Dynamic analysis has been done to get the required time velocity and acceleration of the moving disc.

These mechanisms are fabricated and assembled in the store. Verification and validation trials are completed successfully. Dynamic analysis is carried out for both room and operating temperature (- 40° C).

Separation Dynamics Design for Separation of Moon Lander from Spacecraft in Moon's Orbit

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Separation system using springs are employed as safe separation of lander from a Moon- orbiting Spacecraft is important and crucial task particularly because the lander generally has critical instruments which will operate during its descent. During the separation, there should not be any collision between moon lander and spacecraft. In addition to the collision free separation, the main requirement is to keep the lateral body rate of lander within a narrow & tight specification. One reason for this is to keep the communication alive between lander and spacecraft. Also, if the pitch and yaw rates are high after the separation of lander, the camera field of view may go out of range before any meaningful data is acquired. To start the separation system design, the tolerances on various parameters need to be worked out. An extensive six degree of freedom separation analysis was performed for arriving at an acceptable separation system. The dispersions on various parameters were chosen in line with the dispersions which is normally considered for the separation of spacecraft from a launch vehicle upper stage. The analysis indicated a very high lateral body rate on Moon lander which is not acceptable to the Mission. These high lateral body rates occurred mainly due to the presence of spring force dispersion & lateral c.g. offset together with its dispersion. In addition to the above, the low inertia properties of the lander magnify the effect of dispersions resulting in still higher body rates. Based on the separation dynamics analysis the spring thruster force levels were decided along with their location and all critical parameters were identified and, appropriate values of tolerance levels over them were recommended for the rate reduction. After iterations with the hardware developers, new tolerance limits were finalized. These limits are one order narrower and tighter than those in practice for launch vehicles. Design studies with new tolerances indicate acceptable levels of lateral body rates on the Moon lander. Additionally, the presence of separation plane connectors are essential for system performance. But they cause problems for separation if the spring force levels and their location are not properly designed. A strategy for this was also worked out from the separation dynamics analysis. The exercise ended in the finalization of the separation system specifications for a safe separation.

DM/O/09

Kinematic Analysis of Main Door Locking Mechanism for a Civil Aircraft

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This paper presents the 3D kinematic analysis of the Main Door Locking Mechanism of the 14 seater Saras Light Transport Aircraft. This complex mechanism comprises of 55 components and 66 joints that were modeled and assembled using CATIA V4. The kinematic analysis of the mechanism was carried out using CATIA V4 Kinematics module, based on the initial layout proposed by the Fuselage design group. The primary objective of this analysis was to ensure that all the lock pins move the same lock distance (~17 mm) for a given angle of handle rotation (~35°). The kinematic simulation of this initial layout showed that all the lock pins were moving in different lock distances when the handle is rotated by a given angle, which was undesirable. This necessitated changes in the geometry of the components leading to achievement of the desired objective and enhanced understanding of the 3D kinematics to solve typical problems of this kind. This paper puts forward in detail the different methodologies that were adopted and the modifications carried out progressively to achieve the desired objectives and also present the results thereof.

Effect of Wedge Angle Tolerances on Clamping Force Distribution at Interface Flanges in a Merman Band Separation System

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Merman Band Joint is used extensively for attaching spacecraft to launch vehicle and separate it on command for injection in desired orbit. In this system , the spacecraft interface ring and payload adaptor (PLA) fore end ring of launch vehicle are held together by a set of wedge blocks wound by a metallic band. This system is simple & reliable besides offering excellent joint characteristics.

The clamping force is generated by the wedging action of wedge blocks held by the tensioned band. In order to ensure close matching of interface flanges with wedge blocks and also to obtain a near surface contact between them after tensioning the band , stringent tolerances are specified for these three interfacing elements. As the spacecraft interface flanges and payload adaptor are part of expensive structures, suggestions to modify/ rework will always end up on the wedge blocks even when no deviations are found in the latter.

By the configuration of interfacing members , an interference fit of wedge blocks with interface flanges will lead to a better joint as the clamping force is applied close to the web. However, this leads to high contact stress at fore end tip of wedge block as well as its opening out. On the other extreme , a clearance fit will result in poor joint characteristics as in this case the clamping force is applied at the tip of flanges. The clamping force distribution on the interface flanges is found to be a measure of joint characteristics.

This paper describes the results of Axi-symmetric FE contact analysis carried out to find out the effect of tolerances on wedge blocks towards clamping force distribution between spacecraft and payload adaptor and also the acceptance tolerance specifications arrived at for the merman band joint system of spacecraft.

Study of Micropropulsion system and Microthruster for avionic/space applications

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MEMS technology in many space and avionics application add value simply through reduced mass, volume and required power of components and subsystems. Every spacecraft has a number of sensors onboard. Introducing MEMS technology offers a significant mass reduction from several hundred grams per sensor using conventional technology to tens of grams for a MEMS sensor packaged as a stand alone sensor. The inherently low mass, volume and power consumption of MEM devices add value to any space system in terms of reduced costs and also allows increased redundancy, performance and multifunctionality.

Another aspect of MEMS technology for space/avionics applications is as an enabling technology for new missions. One such example is the miniaturized propulsion system described in detail in this paper. Such a system opens for formation flying or precision control of small and medium sized satellites and micro aircrafts with limited resources in terms of mass- and power budgets. The MEMS based micropropulsion system in principle is similar to conventional cold gas propulsion system but with some difference, that the thrust generated can be modulated proportionally in the sub milli-Newton range instead of on/off modulation.

The aim of this paper is to study a basic micropropulsion system, MEMS based microthrusters and its design challenges. Although the micropropulsion system consists of not only microthrusters but many other subsystems such as microvalves, microchannels, pressure sensors & regulators etc, the review of various MEMS based microthruster has been done in the paper. Finally, a case study of a vaporizing liquid microthruster (VLM) has been reported wherein the design and fabrication of a vaporizing liquid microthruster has been described.

Nozzle Closure System for gsLVM3 Launch Vehicle

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The gsLVM3 of ISRO is a three-stage launch vehicle capable of placing 4ton class satellite in geostationary orbit. The vehicle has two solid strapon rockets (S200), which are fired on ground. The core consists of two liquid engines (L110) that are fired during flight, before burnout of S200 strapons, nozzles of the two air lit liquid engines are exposed to high thermal radiations from S200 plumes during flight. At higher altitudes, the plumes of S200 expand and interact generating reverse flow of hot gases.

A nozzle closure system is provided on the L110 engine nozzles to protect it from the severe thermal environment due to radiation from exhaust plume and to prevent the entry of hot exhaust gas from the reverse flow of S200 plumes. The closure is separated before the firing of L110 engines, using a separation mechanism. The closure is interfaced to the nozzle through sixteen spring loaded latches which are kept in latched position by a pre tensioned wire rope-pulley arrangement. The latching mechanism is designed for loads on the system due to inertia and differential pressure. The adequacy of the design is verified by kinematic analysis of latching mechanism. The thermal insulation of the closure consists of a six layered composite made up of silica sheets and aluminium foils. The composite insulation is supported by a dome shaped metallic framework. The system is separated on command using pyro wire rope cutters.

The air entrapped inside the nozzle divergent is vented out through vent ports to limit the differential pressure to the design value. Since the vehicle has an acceleration of 3g at the time of separation, inertial force is used to give separation velocity to the separated part. This paper explains the system design, modeling and Kinematic analysis of nozzle closure system for gsLVM3.

DM/O/13

Mechanism for Blind Mating of Electrical Connector

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Reliable electrical connection is required to be established and maintained at 12 m depth between connectors mounted on two independent structures in dynamic conditions. The male part of the connector is mounted on the mechanism which is connected to floating body experiencing sea state level conditions. Female part of connector is mounted at bottom of long tube which is to be installed within a cavity into the floating body. The blind connection so obtained at this depth is required to be maintained for sufficiently long time in dynamic conditions.

A mechanism is devised at R&DE(E) to suit these requirements. Since both the bodies are having motions, the mechanism has to cater for misalignment in longitudinal, lateral and angular direction. A locator on the top visible surface ensures and indicates alignment of both structures. The mechanism is designed, developed, validated and used for the critical application and is in regular use.

The complete mechanism is subjected to rigorous qualification at component and assembly level simulating the working conditions. This paper brings out the features of the mechanism including design parameters and qualification testing involved.

Design of Mechanisms for High Rate of Fire Double Barrel Air Defence Gun

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High rate of fire, medium caliber automatic guns are mounted on fighter air-crafts and combat helicopters as direct firing weapons to engage the moving aerial and ground targets. These guns are required to deliver 6 to 25 rounds in extremely short burst time. Different gun mechanisms are incorporated which works in close synchronization and perform sequential operation of functional cycle to meet the said requirement. This paper brings out details of design and simulation studies carried out for proposed 30mm double barrel gas operated gun. The weapon comprises of two barrels housed in a single gun casing within close dimensional and mass constraints. Each of the barrels is provided with its own breech bolt, while the other mechanisms are common for both of them. Gun mechanisms are driven by gas operated system by tapping part of propellant gases from barrel after firing. Both barrels fire alternatively and energy tapped on firing of one barrel is also utilized to drive the mechanisms of other barrel by kinematic coupling of mechanisms. To achieve higher rate of fire it is essential to reduce cycle time either by increasing the velocities of bolt-slider or by shortening its stroke. The higher displacement and velocity of driven member i.e. breech bolt, is achieved by interfacing the accelerating mechanism with driving piston-slider.

This paper deals with theoretical solution for accelerating mechanism which involves obtaining displacement-velocity profile of slider and breech bolt throughout the cycle and finalizing control profile of the cam. This has got direct bearing on cycle time, rate of fire as well as quality of automatic weapon functioning. Based on the initial inputs from solid modeling, CAD layouts, cyclograph and pressure-time profile of barrel, simulation model have been developed. To avoid sudden changes in motion of driven members, continuously varying transmission function for control profiles of cam have been selected. Output data in respect of displacement-velocity profile w.r.t. time directly gives the time of functional cycle. It is found that rate of fire of 1000 round per minute per barrel is feasible for proposed gun with the designed mechanisms and selected data. Sensitivity analyses have also been carried out to study the effect of variations in spring stiffness and driving-driven masses. This simulation gives overall idea of functioning of weapon mechanisms and provides basis for finalization of critical dimensions of gun mechanisms for achieving desired cyclic rate of fire.

DM/O/15

Auto Balancing of Recoil Operated Guns

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In most of the medium calibre guns, automation is achieved by utilizing the recoil energy of the gun. During firing, the barrel assembly moves back and forth against a recoil spring and buffer.

Due to space constraints, the elevation axis (trunnion axis) of the gun is generally kept behind the centre of gravity of the elevating mass resulting into static unbalance about the trunnion axis. This causes more manual effort for gun elevation and increased dead load on the elevation drive. Movement of the recoiling mass during firing, generates dynamic unbalance. This increases the dynamic load on the elevation drive and reduces gun pointing accuracy. The static and dynamic unbalances vary according to the gun elevation angle.

This paper discusses in details about a unique mechanism evolved to minimize the static and dynamic un-balances. This helps in reduction of manual effort in elevation of the gun and reduction in static load on the elevation drive. It also improves the gun firing accuracy by instantaneous compensation of the dynamic load. In this mechanism the counter moments are generated by pulling the elevating and recoiling masses through the force developed by spring and hydro pneumatic source. Dynamic analysis has been carried out to optimize the location of input points and magnitude of the forces.

This mechanism has been implemented in 40 mm automatic gun for Infantry Combat Vehicle in which the static unbalance was 2000 Nm and the dynamic unbalance was 500 Nm at 0 deg gun elevation angle. This resulted into development of two compact, maintenance free and rugged balancing gears namely static and dynamic balancing gears. The static balancing gear uses Hydro-pneumatic device and dynamic balancing gear uses Mechanical spring. The performances of the balancing gears were verified during firing of the gun.

Design And Analysis of Inlet and Exit Cowl Actuation Mechanisms For HSTDV

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DRDL is developing a Hypersonic Technology Demonstrator Vehicle (HSTDV). CADES has carried out design and analysis of the inlet cowl and exit cowl actuation mechanisms for DRDL, The inlet cowl actuation mechanism is to provide sufficient flow of air during the engine starting phase. The exit cowl is to provide aerodynamic stability of the vehicle during the power off condition. Deflecting the cowl of HSTDV for a brief period is very critical from pitch stability. Considering the various constraints such as low volume, less weight and high temperature it is required to design a robust mechanism to deflect the cowl.

Inlet cowl plate is initially at 27.8° (closed condition) and the inlet cowl plate has to rotate about hinge to final position (open condition) within a specified time interval of 100 milli seconds. The side links of the mechanism are routed through a small space available in between the bi-wall of the engine. No leakage of gases is allowed through the gaps between the cowl and sidewall. The mechanism to actuate is designed for overcoming the predicted aerodynamic loads, inertial and frictional forces of the inlet cowl plate. This mechanism is also designed for the thermal stresses due to the thermal loads.

The exit cowl mechanism designed is intended to deflect the cowl plate by 25 degrees in 100 milli-seconds and bring it back to the undeflected position. This is achieved through the use of a linkage mechanism. An actuator is used to provide the required movement to the links. To avoid obstructions to the flow area, the side links of the mechanism are routed through the space in between the bi-wall of the engine exhaust. In the areas, where there is exposure to high temperature, appropriate material is used for the links. The mechanism, chosen from several concept designs evolved was tested in multi body dynamic analysis package (MSC ADAMS) to obtain the forces acting on the mechanisms. These forces were used to carry out the FE analysis to test for the strength of all the components of the mechanism. The actuation forces were also obtained from MBD analysis. This mechanism was checked for functionality under the prescribed loads. Feasibility of manufacturing and assembly was also kept in mind during design. Reliability and tolerance studies were also carried out. The results of the MBD analysis for the inlet and exit cowl mechanisms are presented in this paper.

DM/O/17

Prediction of Shock Levels in a Clamp Band Separation System : A Mathematical approach

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Clamp band separation system is used for attaching satellite to launch vehicle and dispense it on command. In this system, the satellite adapter and payload adaptor (PLA) are held together by a set of wedge blocks wound by a metallic band. The band is made in two segments and connected by bolts. The joint is preloaded by tensioning the band through tightening the bolts. Separation of satellite is achieved by severing the bolts using pyro cutters.

In this system, shock generated during separation is mainly due to the sudden release of strain energy stored in the interface flanges while tensioning the band. To estimate the peak shock generated during separation, a mathematical model based on Lagrange's equation for dynamic systems is generated.

This paper describes the formulation of mathematical model for estimating shock generated during separation for various pretensions of band and its validation using shock data measured in ground test. A close match is observed between prediction and ground test results.

Design Optimisation of Single Rupture XTA System for Futuristic Application in Stage Separation of Sounding Rockets

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Separation Systems based on Expanding Tube Assembly (XTA) finds its application where there is a requirement of synchronous linear separation with full confinement of detonation products contributing minimum level of shock. For its unique advantages, it is generally employed for application in the upper stages of launch vehicles viz. separation of payload fairings, dual launch adaptors, micro satellites etc. XTA based separation systems essentially consists of a structurally weakened section with a stress raiser configuration, which gets severed during expansion of adjacent flattened tube under explosive stimulus. Single as well as double rupture configurations have been successfully developed for various applications.

A single rupture configuration of XTA based system which was demonstrated successfully at 2800mm diameter ring level was considered for its usability for a futuristic application in 560 mm diameter sounding rocket towards separation of booster and sustainer. Sounding rocket application demands higher cutting thickness to meet flight loads at separation plane and the single rupture configuration was chosen due to its maximum cutting thickness for the same explosive charge quantity. A partial severance of the 560mm dia ring was noticed during development even though similar configuration functioned successfully at straight panel level as well as at 2800mm ring level. Cracks were noticed to initiate from internal notch and propagate in a direction different from the external notch, unlike the pattern observed in higher diameter ring.

At this juncture, a detailed finite element analysis was carried out to optimise the existing configuration in order to efficiently utilize the available explosive energy without sacrificing the margins on flight loads. A 2-D axisymmetic model is taken for the finite element analysis to study different notch locations in order to achieve better severance pattern under minimum pressure inside the tube. A 3D model of the ring at critical location of each configuration is considered to estimate the margin on flight loads. Based on the analysis , the final configuration was arrived at optimizing both requirements of higher margin on flight loads and complete severance with available explosive energy. A functional test at full scale ring level was carried out subsequently in the finalised configuration demonstrating complete severance.

This paper describes about the details of structural analysis and the functional tests carried out.

DM/P/19

Thermo-Elastic Analysis of A Dual Gimbal Antenna Pointing Mechanism

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AThe details of a thermoelastic analysis on the Dual Gimbal Antenna (DGA) pointing mechanism of a typical spacecraft intended for interplanetary/lunar mission has been presented in this paper. The DGA mechanism on these missions helps to orient the antenna continuously towards earth station for data transmission, while the spacecraft orbits around the planet/moon. Typically, two orthogonal drive modules steer the antenna in two axes. To offload the launch acceleration loads / propulsion engine loads to hard points on the support structure and to avoid excessive load transmission to the drive module bearings, the antenna is kept stowed during launch and deployed when the spacecraft reaches the requisite orbit. A schematic of a typical DGA system on spacecraft is shown in Figure-1 below.

Before release, the DGA system is subjected to varying thermal environment due to different spacecraft orientations in the transfer orbit and in the final intended orbit. While orbiting around planet/moon, the spacecraft faces temperature gradients depending on the sun angle and its position in the orbit.

Thermal stresses are developed in the system, since it is a constrained system subjected to temperature variations with respect to time. The thermal stresses are a function of different factors viz., the thermal expansion coefficient, modulus of elasticity, temperature gradient, temperature difference with respect to the ambient ground temperature and the nature of the constraints. The present study aims at simulating the varying thermal conditions on the system in orbital geometries and to determine the resulting thermal stresses.

Typical spacecraft orientations have been identified for analysis. For example, in the dawn-dusk orientation the temperature gradients are constant with respect to the position of spacecraft in the orbit. However, it changes continuously for the noon-midnight case. The temperature distribution in any other intermediate position would be within the temperature extremes defined by these two orientations.

A mathematical model of the DGA system has been developed. The thermal implementation on this model provided the typical temperature distribution data. This data has been obtained for thirteen different configurations for noon-midnight orbit and one configuration for dawn-dusk orbit.

The temperature data has been converted to NASTRAN readable format using an application specific C program. Structural analysis on the DGA system has been carried out imposing this temperature distribution data. The constraints among different constituents are defined using MPCs (Multi Point constraints). Figure-2 presents the mathematical idealization of the system under study. Fourteen different

case studies have been carried out simulating all the different temperature distribution conditions.

The results from analysis include determination of stresses due to the imposed thermal loads. It is found that the stresses are within limits. The bolt joints at different interfaces have been analysed separately and the stresses have been found out. The mechanism design incorporates two sets of bearings and it is necessary that thermoelastic analysis is required to be carried out to make sure that these bearings do not get jammed/over loaded and sufficient clearances are ensured. The axial and radial loads at all the bearing locations have been determined.

The present paper provides the details of modelling, results and discussions.

Modeling and Simulation of Fin Actuation System for an Antitank Missile

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The Fin Actuation System (FAS) deflects the control surfaces projecting out of the missile body so that resulting aerodynamic forces act on the missile. These forces manoeuvre the missile towards the target and stabilize it against any external disturbances. The FAS consists of a servocontroller, motor and a transmission mechanism along with the fin. Thus the system is highly multidisciplinary in nature. The response of FAS to a control command attributes in deciding the trajectory of the missile and the nature of missile-target interaction.

The linkages of transmission mechanism may fail due to constraint forces acting on them. Therefore, an effort has been made to develop the model of Fin Actuation System and analyze it to find out the dynamic response of the system and the constraint forces. The servocontroller and motor is modeled in MATLAB/Simulink software and the mechanism in ADAMS software. The model of transmission mechanism was built in ADAMS and later exported to MATLAB so that control system built in MATLAB/Simulink interacts with mechanical system built in ADAMS in real time, thereby modeling the compound disciplines involved in FAS.

A step of 15° fin deflection from 0 to 1 second was input to the system and the response was obtained. The reactions on all joints in the mechanism were obtained and strength of screw threads was checked. To validate the results of above simulation, complete mechanical system was modeled in MATLAB/Simulink using mathematical equations along with the control system already built while creating its interface with mechanical system in ADAMS! It was simulated to obtain some key results. The results match fairly with combined MATLAB/Simulink and ADAMS simulation thus validating the ADAMS simulation.

Regarding the behavior of the system, slight overshoot is acceptable with a view to obtain fast response and effect of required fin deflection. The mechanism was found safe for power transfer. The system model developed also evaluates the performance of the PID servocontroller and if needed, the PID gains can be instantly modified to obtain required system response characteristics. This way, the model may be useful in modifying or optimizing the design of the PID controller. The response of the system to the control command can be visualized in soft environment. Through this work, an interactive, software based technique for modeling, simulation and analysis of such a multidisciplinary system has been demonstrated.

The scope of this paper entails linking the control system software like MATLAB/Simulink with mechanical system software like ADAMS so that input to the transmission mechanism i.e. driving torque is exactly modeled and mechanism can be analyzed correctly. The detailed analysis of control system and motor along withtheir electronics is beyond the scope of this work. There are many industrial applications in which a mechanical system is driven by a control system. This paper presents a relatively new, CAD/CAE based technique for design and analysis of such complex, multidisciplinary systems.

Graduating from Conventional Safety Arming Mechanism (SAM) to Micro Electro-mechanical System (MEMS) Based SAM

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A conventional Kill Mechanism in any form whether, as a part of an Ammunition, a Missile or Rocket Warhead or a Mine, requires a device called the Fuze which broadly controls and governs the characteristics and role in which the kill mechanism is used. The Fuze is a device incorporated with necessary safeties and explosive elements designed to initiate a train of explosives leading to detonation of the Kill Mechanism at a predetermined point in time and space. It renders all the safeties to the kill mechanism during assembly, storage, handling, transportation and till the terminal phase of its delivery to the target end. Rightly has been named as the brain of the system by the Armament designers.

The present trend in armament technology is to graduate from the so called "Dumb" Ammunition to "Intelligent", "Smart" and "Brilliant" ammunitions. The main contributor in this process of transformation is the Fuze. A fuze has four major sections namely, Electronic Section (Brain), Safety and Arming Section (Brawn), Power Section and Explosive Section. The topic of discussion for the current paper entails us to concentrate and highlight on the importance of the Safety and Arming section of the Fuze only.

As the name suggests, the purpose of the Safety Arming Mechanism (SAM) is to guarantee safety to the ammunition throughout its life cycle i.e. safety during manufacture, handling, transportation, storage and during flight till the predetermined point is reached. Generally two independent safety interlocks utilizing two independent environmental forces for their removal are employed in the SAM. Accuracy, repeatability of operation and reliability of functioning are the hallmarks of the SAM design. Necessary redundancies are also incorporated for critical/crucial operations of the SAM in the design. Finally, the SAM has to meet the size, shape, mass and volume requirements stipulated by the Kill Mechanism designer. After incorporating all the above requirements, the SAM becomes an assemblage of a variety of small and intricate mechanical components of diverse materials, springs, rotating and sliding devices, mechanical linkages, actuators, sensors, explosive trains and built-in redundant mechanisms. Electrical interface issues with Electronic section and mechanical interface issues with the Kill Mechanism complicate the design further. Ensuring a very high repeatability and reliability of operations with such kind of an assemblage is a Herculean task.

To reduce the size and cost and increase the accuracy, consistency and reliability of the Fuze (SAM), MEMS based components of late have become inevitable. Due to its smaller size and mass, gravity and inertia become insignificant whereas atomic forces and surface science become highly important. The smaller size, low cost and higher reliability of MEMS based devices have made them the first choice of designers for all futuristic Fuze designs. In this paper, efforts have been made to present the necessity and advantages of incorporating MEMS based devices for Fuze and SAM applications.

Design and Development of Mixing Reactor for Bio-mimetic Synthesis under Micro gravity on-board first Indian Space capsule Recovery Mission

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There is considerable interest worldwide to conduct microgravity experiments in space and making them commercially viable. Due to unique condition offered by space, there are many processes, materials and products which are possible to make in space which otherwise are not possible to create on earth. For example life saving drugs (inter-pheron for cancer treatment), bioactive products, perfect crystal, perfect sphere and container-less heating etc. Many space faring nations conduct microgravity experiments on unmanned recoverable satellite systems. Indian Space capsule recovery mission, which successfully recovered micro gravity experiments, was one such attempt in this field.

The paper describes the design details of this module considering the constraints such as mixing of two reactants only after reaching the orbital micro gravity condition, reliable actuation and ensuring mixing under micro gravity, and no leakage under space environment. A pyro-operated mechanism with glass as the separation wall was conceived and built. Provisions are made for pyro-cutter actuation onboard by pyro firing package and telemetry of actuation signal to ground.

The reactor is successfully used for bio-mimetic synthesis (payload) of hydroxiapatite nanoparticles under micro gravity environment on ISRO's Recovery Mission (Figure-2). The objective of this experiment was to study the biomimetic synthesis of hydroxiapatite nanoparticles under micro gravity environment. For this space processed sample in mixing reactor was planned to compare with ground experiment results. In this experiment a liquid diammonium hydrogen phosphate and Ca-ion loaded PVA-collagen mixture (a bio-mimetic gel) are placed in the reactor and experiment is conducted by actuating mechanism. The study of space processed samples received from reactor after recovery of S/C suggests,

a. Almost all the liquid and gel are reacted i.e. full volume of sample is processed and reaction is completed.

b. The morphological features through out sample were similar.

c. These features are founddifferent from that obtained from ground experiment.

This paper highlights realisation of a mixing reactor and the glass breaking mechanism to allow the mixing of two liquids or liquid with semi solid/ gel for an unmanned recovery satellite will be discussed. The paper will briefly discuss the experiment objective & its on-orbit performance.

Design and Analysis of Pyro Lock Mechanism for Nag Missile

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Missile will be kept in Launch Tube / Canister during Transportation. In order to avoid the movement of the missile during Transportation, a locking arrangement is required. A pyro operated lock pin mechanism has been designed to arrest the movement of the missile in the Launch Tube / Canister during transportation (in the captive mode).

Pyro Lock Mechanism mainly consists of three components namely piston, pyro body and flange, driven by pyro cartridge. Piston serves as a locking pin. Pyro body houses the piston and pyro-cartridge which moves the piston. The mchanism is mounted on the other canister by means of a flange. Piston holds the pyro cartridge at its head and the other end of the piston (piston pin) will engage into the cavity of the missile section. The piston and the pyro-body are locked in position by means of a shear pin. High-pressure gas sealing is provided by means of 'O' ring and gasket. When pyro is ignited high pressure gas will be generated inside the piston cavity at its head. This gas will pass through the 4 holes in the piston and enter into the space between the body and the piston and pushes the piston upwards which breaks the shear pin. When the piston is moved up by 7.5 mm, the missile is unlocked from the canister and ready for launch. As the piston moves up and reaches the step the circlip expands and locks the piston at that position so that it will not come down.

This paper deals with the Classical Design and Finite Element Analysis of the Pyro Lock Mechanism.

Study of wrap-around system of propellant feedlines for gimballing of a liquid rocket engine

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Gimballing of a rocket engine is one of the methods of achieving thrust vector control in order to correct the deviations in the trajectory of the rocket. In a liquid rocket engine metallic feed lines are provided to supply liquid propellants from the tanks to the engine one each for the fuel and oxidiser. It is required to ensure adequate flexibility to these feed lines so that gimballing of the engine is done without any resistance. One of the popular methods of imparting flexibility to these lines is by providing gimbal bellows- usually three per line.

This paper discusses the structural adequacy of a wrap-around system of feed line with gimbal ducts or gimbal bellows for a liquid cryogenic rocket engine. The rationale of providing the wrap around system is initially explained. Details of the above configuration are also highlighted. Subsequently kinematic analysis of the ducts done through finite element analysis, to ensure adequate flexing of the engine, is discussed. Structural analysis is carried out using ANSYS (version 11.0) general purpose FEA code implemented in a high performance graphic workstation.

In a wrap around configuration, the feed lines are wrapped around the rocket engine to minimise the space consumed and also to minimise the weight. There are two feed lines in the system; one for the fuel and the other for the oxidiser. Three gimbal expansion joints (gimbal bellows) are provided in each feed line along three orthogonal axes to provide sufficient flexibility. Gimbal expansion joint consists of two gimbal cups kept orthogonal and connected through a ring kept inside with the aid of two pairs of pins. The gimbal expansion joint is connected to the feed line through welded joints. The fluid flow inside the feed line and the joint is permitted by providing a flexible bellow inside the ring of the gimbal expansion joint, which provides rotational degree of freedom about two transverse axes.

The angular rotation of each bellow during engine gimballing is estimated and checked for its adequacy. The other analyses carried out are

1. elasto-plastic stress analyses of gimbal duct assembly for different load conditions
2. estimation of natural frequency of the system
3. harmonic response analysis of the wrap around system.

Detailed FE analysis is carried out for a single gimbal duct assembly. Stresses are estimated for the operating and proof load conditions. Configuration of the gimbal cup is optimised by changing the geometry and thickness at certain locations keeping the condition of 'no yielding at proof pressure load'. The burst pressure of the gimbal expansion joint is also estimated. Bi-linear kinematic hardening rule employing

von-Mises yield criterion is used for the estimation of burst pressure. Adequacy under vibration environment is checked through natural frequency analysis and harmonic response analysis. Results are presented in graphical form and concluded that the present configuration is structurally adequate with sufficient flexibility for the specified pressure loads and vibration environment.

Induction of Mems Inertial Sensors for Space

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Inertial Sensors, such as accelerometers and gyroscopes, are widely used in many applications in the aerospace, military, automotive and marine industries. The miniaturisation of inertial sensors have made been possible by advances in silicon microfabrication technology and the emergent field of MEMS sensors has grown rapidly during the last two years with a mature technology and high volume production at low cost. Now, they are in the best position to successfully move in to traditional inertial applications and also to render possible newly emerging applications. The domain of application of MEMS inertial sensors can be classified in to two distinctive types: Movement detection and Measurement and Control. Movement detection is largely driven by automotive industry and low cost consumer business. On the other hand, Measurement and Control is driven by the flight stabilisation and inertial navigation requirements of aerospace and defence sectors. Conventional inertial sensors such as electromechanical products, historically served this domain and due to its high cost it cannot penetrate beyond the realm of aerospace and defence sectors.

The first inertial sensor to come to fruitation using MEMS technology was accelerometer and they are the stars of the MEMS show with variety of products and its extensive usage in automotive industry. A typical MEMS accelerometer measures the component of translational acceleration minus the component of gravitational acceleration along its input axis. A gyro measures angular rotation with respect to inertial space about its input axis. The sensing of such motion could utilize the angular momentum of a spinning motor, the Sagnac effect on counter propagating light beams or the Coriolis effect on a vibrating mass. MEMS gyroscopes to date operate on the principle of detecting induced coriolis acceleration on a proof mass that vibrates along an orthogonal direction to the applied input rotation. As such, MEMS accelerometer technology is in a matured state, but MEMS gyro development is trailing behind. as they are structurally and electronically more complex. But recent new gyros shows promising trend of providing better performance.

As per the present state of technology of MEMS inertial sensors have low precision compared to navigation grade classical inertial technology. In theory, the basic sensor limits are drift stability and random walk. Performance levels achieved range from automotive to tactical grades and lags far behind high performance required for Measurement and Control domain of aerospace industry. But, they offer alternative opportunities toward achieving stringent accuracies by GPS aiding, error modelling and real time error compensation techniques using advanced estimation techniques such as kalman filtering. This paper discusses on various issues related to MEMS inertial sensors and their applications to the stringent navigation grade requirements of Measurement and Control domain of aerospace.

Fragment Generator Velocity Estimation

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Neutralization of Tactical Ballistic Missile (TBM) class of targets has been a challenging task to the weapon designers due to their hypersonic velocities and low Radar Cross Section (RCS). Further, TBM generally carries warhead of mass destruction. i.e. Nuclear, Biological, Chemical (NBC) etc. Hence, TBM target needs to be neutralized in their exo-atmospheric flight, so that in any eventuality, the effects of the warhead would not reach the ground. With the technologies available, scoring of direct hit of these targets in their flight is a rare situation. Hence world over, such targets are neutralized by a fragmentation warhead which can concentrate a high density fragment beam (> 50 hits/m^2). Out of the several fragmentation warhead options available, the Fragment Generator (FG) warhead is generally preferred due to its simple in construction and reliable in functioning.

FG warhead consists of a multilayered fragment matrix disc and a high explosive column behind it. The warhead face is oriented towards the target in the final engagement scenario and initiated. On its detonation, the warhead projects large number of fragments toward target. In order to neutralize the TBM class targets, the placement of fragment beam of the warhead on the target is critical due to very high closing velocities and low RCS. Hence precise estimation of fragment velocity is essential.

Few formulae have been published in the literature for estimation of fragment velocity for FG warhead. These formulae take into account C/M, L/D and Tamper Mass effect. However, the estimated velocity is not in good agreement with experimental results. Moreover, no mathematical formulation is available to predict layer wise fragment velocity of multilayered fragment matrix and effect of charge configuration.

Authors have conducted experimental trials & simulation studies to develop mathematical relations for estimating the fragment velocity, taking in to account the effect of C/M, L/D, Tamper Mass & Charge Geometry together. Also, a relation for estimation of layer wise fragment velocity of a multilayered fragment matrix has been developed. This paper brings out the details of studies carried out and the mathematical relations evolved for fragment velocity estimation.

DM/P/27

Design Analysis of Thin Metallic Diaphragm used in PYRO Devices for Low Temperature Applications

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Pyrotechnically actuated devices find major applications in launch vehicles and satellites for performing various critical functions at different environments. The baisc working principle of these types of pressure actuated devices is the piston cylinder mechanism. Here, a piston is pressurized by pyro gases generated by the combustion of Pyrotechic charges and this in turn drives the piston forward to do the intended function. Being pressure dependent systems, sealing, both static and dynamic plays an important function in these devices. The sealing requirement is not only for preventing the gas leak from the pressure acting volume, but also for preventing the contamination of the combustion propducts with the working medium or environment as per the functional requirement. In most of the devices meant for normal use, O-rings are extensively used to achieve the sealing requirements. But the choice of O-ring for these applications is limited by the operating temperature range of these devices. For extremely low temperature range that is for devices meant to work under cryogenic environments the sealing design assumes further dimensions. Here, O-rings cannot be used as they loose their sealing effectiveness, at low temperature and undergo glass transition.

Here, the flexibility and diversity of thin metalic diaphragm are made use of to obtain effective sealing solution for devices for low temperature use. Thin metallic diaphragms are configured and interfaced suitably for static and dynamic sealing purpose. They are also used in the form of flexible actuator elements in piston cylinder arrangements in pyro devices for low temperature application. In this design, pyro gas pressure pushes the diaphragm to enable a linear motion while maintaining necessary sealing. Interestingly, the functional requirements of the diaphragm are contradicting they should be sufficiently strong and should withstand the pressure, as well as they should not be too thick so that it affects its flexibility. A thorough understanding of diaphragm characteristics is essential as the sizing, thickness and configuration is narrowed down in the initial design phase itself even before realization effort for the diaphragm is started. It may also be noted that the tooling requirement for realizing the diaphragm are expensive and time consuming and further any subsequent change in the diaphragm are expensive and time consuming and further any subsequent change in the diaphragm will not be possible with the already realized tooling. This paper highlights the attempts made to predict the behaviour of thin metallic diaphragm at different loading condition by Finite element method of analysis. The approach includes analytical methods, Finite Element Analysis and experimental validation.

DM/P/28

Challenges in the Design and Development of a Pyrotechnic Parachute Release Mechanism

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Various mechanisms can be employed for holding a deployed parachute and realsing the parachute after its intended purpose. For critical applications, which require high realiability, pyrotechnic release mechanisms are used. These are compact in size with high load capabiliy and can perform the function within milliseconds. Parachute releaser is a pin puller type of device. The pyro mechanism acts by retracting the load-bearing piston, which protrudes into the body of the device thereby realising the load attached to it. The extended piston passes through a slot provided in the body, through which an attachment is provided for connecting the parachute bridle. A shear pin arrests the initial movement of the piston. When the piston is retracted into the body of the releaser, the load path is disconnected and the clamp attachment of the parachute is free to move out of the device. These types of devices are compact for accommodating in the samll space available and optimally designed to withstand the functional loads.

The challenges involved in the development were the choice of material and configuration for the high load capability within the specified envelop, sealing requirement, evaluation of the structural and frictional loads for the bearing surfaces for the release pressure estimation, demonstration of under water functioning performance, devising a suitable setup for functional testing, locking mechanism for preventing the rebound of piston after ejecting the clamp etc.

The releaser design was carried with the above special features to meet all the design requirements and adequate number of units was tested to demonstrate the performance in stand-alone mode as well as in system level. Finally releasers were successfully used in Space Capsule Recovery Experiment of ISRO.

Reaction Mechanisms for Canister Launched Missiles

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Advanced supersonic cruise missiles and underwater launched ballistic missiles ejected from a canister by gas generator (GG) and subsequently orientation corrected for mission requirements demands short duration, multiple low pulse thrusters. These short pulse solid thrust motors required to manipulate the reaction control system consists of Roll stabilization motor, Roll correction motor, Pitch turn motor and Separation motor. Gas generators consist of launch gas generator for ejection of missile out of canister and a cool gas generator for unfolding of wings after missile come out of canister. To keep the technology of reaction mechanisms abreast with world scenario a wide variety of extruded propellants with highly reliable pyro and ignition systems have been developed.

The mechanisms consisting of (a) high burn rate Extruded Double Base (EDB) propellants for different short pulse thrusters and gas generators (b) suitable igniter and (c) pyro initiation system, with adequate no fire capability to encounter EMI and EMC problems. Short pulse thrusters are used to generate required thrust in stipulated time. The parameters like pressure-time and thrust-time are required to be critically controlled for reliability of desired operation. GGs are used to generate required working pressure inside canister / unfolding system of wings. The parameters like maximum pressure inside canister / wing unfolding system, time for maximum pressure, wing-unfolding time are required to be critically controlled for reliable operation of missile launch / wing unfolding mechanism. For short pulse thrusters accurate thrust measurement is highly skilled job with minimum thrust losses which otherwise occur due to various frictional forces and thrust bed losses. As the testing of short pulse thrust motors having different cant angle nozzles is tedious, special thrust measuring device is required for this purpose. The performance parameters like C*, Isp, burn time of short pulse thrusters are derived from pressure-time and thrust-time profile by static testing in a special thrust measuring device. The performance of the GGs was measured by testing GG cartridges in vented vessel mode and closed vessel test mode. In the vented vessel mode GG is connected to one cc vessel of suitable vent. In the closed vessel test the GG is connected to a known volume simulating the unfolding system volume and the parameters like maximum pressure (P_{max}), time to P_{max} and the ignition delay are measured.

The short pulse thrusters after evaluation by static testing and gas generator cartridges after evaluation in closed vessel and vented vessel were used in flight trials. The mechanisms have performed to the desired levels of accuracy.

DM/P/30

Cryogenic Umbilical System for Launch Vehicles

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Liquid propulsion stages of launch vehicles require an inevitable system known as "Umbilical". The system essentially consists of the functional sub-assemblies namely flight segment, ground segment and lock & separation mechanism. The unit acts as an interface between launch complex and the liquid stage in order to facilitate operations such as cryogenic propellant filling & draining, helium command gas supply for pneumatic actuation of control valves, ullage volume of propellant tank charging & venting with helium gas, supply & release of helium purge gas for cryogenic engine system, charging & venting of high pressure helium gas to storage gas bottles, supply of gaseous nitrogen to fire & safety system etc. in addition to enabling the cryogenic fluid servicing at launch pad, the unit has to separate the ground segment at the time of lift off of the launch vehicle. The cryogenic stages of ISRO's launch vehicle employs two umbilical units for the stage fluid servicing operations at launch pad, one called as Fuel Umbilical Unit (FUU) and other Oxidiser Umbilical Unit (OUU). The cryogenic umbilical is a complex system, as it has to operate under extremely low temperatures (20 K), extreme thermal gradients (300 K - 20 K), stringent leak tightness of fluid lines ($\leq 1 \times 10^{-3} cm^3/s$ of GHe at 20 K) and safety & reliability standards. The specification of such a system calls for conceiving, designing and developing reliable mechanisms, fluid transfer lines, material selection, thermo structural analysis etc.

Design & proto assembly of these units were completed and extensive testing is planned to prove its reliability before inducting into launch operations. A series of tests are planned on these units during development, qualification and flight acceptance phase. The paper will touch upon the design features, operational aspects and description of the various mechanisms, actuators used in the cryogenic umbilical system first time developed in the country.

Design of Stabilization System for Ship Communication Application

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This paper highlights the salient features of a new three axis stabilization system, which is aimed to establish radio frequency (RF) communication between two ships in sea environment, which are far away from each other The ship is a floating vessel in marine environment, which are contineously subjected to varying pitch, roll and yaw motion by sea weaves and wind. The amplitude of the pitch and roll movements depends upon the size and stability of the ship vesel and sea state. The developed system stabilizes two RF antennas against the sea water wave distriburance in real time. The sea wave disturbances are sensed by Micro Electro Mechanical System (MEMS) based tilt sensor and feedback signals are send to a Digital Signal Processor (DSP) Controller for correction to pitch and yaw plane by means of external power source. The DSP Controller works in close loop with actuators, feedback devices and tilt sensor. The design of this system has been done with three very stringent parameter like weight, overall size and storm wind condition in mind.

In order to meet the objective, two versions of the system for different configuration have been conceptualized and developed. It is assessed that an applicable performance improvement is achievable by new system. The system is mounted on the mast of the ship at 10m above the water level and control by the controller housed inside the cabin of the ship and controller by the computer.

The system has been evaluated first on ground by mounting on two axes platform simulator. External commanded disturbances are applied to the system and outputs of the systems are monitored by 16 channel recorders. Input and output results are plotted by using MATLAB software. It is found that input and output results are matching, which confirms the validity of the design.

The position of the antenna and orientation is being displayed on the computer screen in real-time. This paper also describes the functioning of both versions of the mechanism.

DM/P/32

Design of High Stiffness Gimbal Suspension Base Structure for Calibration of Inertial Guidance System

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This paper deals with the active vibration control of three point support base structure. The structure is designed to support high power Electromagnetic servo drive system. The cast Iron SG grade 600 material is selected for high damping characteristic. The base structure is supported with ribs to ensure high stiffness and high natural frequency. The dimensional stability of the structure is very critical to ensure wobble and orthoganality of the drive system. Hence it becomes critically important to understand the behaviour of structure in dynamic environment.

A finite element model based on Euler Bernoulli beam theory which is the classical formulation for beams has been used. The structure performance has been evaluated out by applying different type of loads such as impulse, step, harmonic using FEM analysis software. Different modes of natural frequency, twisting mode, maximum deflection and stress were found out by analysis.

The structure performance is a validated by experimental testing. The structure is subjected to actual vibration by suspendng and exciting by external source. Five sets of accelerometers are fixed at different locations to monitor natural harmonic mode of the structure.

Comparing the modal frequencies validates the behaviour of the structure and mode shapes. The results are tabulated for theoretical and practical analysis and found good agreement. The fundamental frequency of the structure is found to be 125 Hz with associated twisting mode of the structure followed by the second mode at 127 Hz.

DM/P/33

Aerodynamic Parameter Estimation using New Filtering Technique Based on Neural Network and Gauss-newton Method

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A new parameter estimation method based on feed forward neural network is proposed. The proposed method uses black box approach to build the flight dynamics model of an aircraft. Gauss-Newton method is then used to obtain optimal valaues of the aerodynamic parameters by minimizing the error cost function. The method has then been validated using flight data pertaining longitudinal dynamics of aircraft. Proof of match approach has bee followed to verify the model estimated by the proposed method. The results obtained using the proposed method have been compared with those obtained using wind tunnel, maximum likelihood and filter error method. Unlike, most of the conventional methods, the proposed method does not require a prior description of the model It also bypasses the requirement of solving of equations of motion. This feature may have special significance in handling flight data of an unstable aircraft.

Neural Networks Based Approach to Model Lattice Fin Aerodynamics

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Aerodynamics of grid fins, operating at high angle of attack is of paramount importance for modern day missiles. Many methods and techniques exist for successfully analyzing such flow for low to moderate angles of attack. However, for high angles of attack, there is a great need for methods/techniques to predict or compute the aerodynamic parameters. The present paper highlights the utility of new approach based on Feed Forward Neural Networks to model lattice fin aerodynamics. Using wind tunnel test data of lattice fins, neural network approach has been shown to have distinct advantages. Once the neural model is established, it could be used to reduce large number of separate testing at different angle of attack. Further, it is also highly effective in capturing aerodynamic model in the nonlinear range of angle of attack. The derived neural model can advantageously be used for control applications. The proposed approach bypasses the requirement of postulation of a priori aerodynamic model. A black box model can be developed using measured motion and acceleration variables.

Actuation Mechanism of Control surface for an Unmanned Air Vehicle

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Control surface actuators are the key systems in any flight vehicle. They drive the control surfaces and position them continuously as per their command signal and counter the hinge moments on the control surfaces due to aero dynamic forces, thus enabling a strict control on some of the flight parameters. A Flight control system consists of an Actuator mounted at a suitable place and connected to the control surface directly or through a linkage mechanism.

Electro Hydraulic (EH) Actuators can provide excellent performance over a wide range of auction power requirements. These Actuators are almost mandatory for high power, high bandwidth operations. However a distinct drawback EH Actuators is the high cost, weight, sealing problem etc.. Electro Pneumatic (EP) and Electro Mechanical (EM) Actuators offer advantages of low cost and less logistics while satisfying the control requirements for low power, low bandwidth applications. EP Actuator with a self contained servo actuation system using a gas bottle to store compressed gas will be ideally suited for a short duration application. EM Actuation is currently an alternative to EH&EP actuation ideally suited for Unmanned Air Vehicle (UAV) applications.

Rotary Actuators normally consists of a servomotor driving a gear train associated with a position feedback and a controller, to position the control surfaces/canards. Rotary Actuators are simple in construction and these actuators can meet considerable accuracies of positioning the control surfaces. However linear actuators with BLDC motors driving a Ball or a Roller screw with linear positional feedback and a controller offer distinct advantage over rotary actuators in terms of position, free play, stiffness and inertia.

This paper deals with design, development and testing of a linear actuator to actuate the canards of an UAV. A Linear actuator using a BLDC motor, LVDT position feedback and a roller screw has been developed to achieve this objective. A crank and slotted lever mechanism has been used to convert the linear motion of the actuator to the rotary motion of the canard. A neutral locking mechanism operated by lanyard has been developed to ensure the locking of actuator before they are powered on.

The actuation system has been subjected to various tests and it has complied very well with the design specifications. Linkage Analysis and Frequency Analysis has also been carried out to ensure the capability of the system. Particular thrust has been given to ensure high overall stiffness and accuracy while maintaining a low weight and inertia.

DM/P/36

Design, Fabrication, Testing and Characterization of Magnetorheological Fluid Based Damper

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Magnetorheological (MR) fluids belong to the class of controllable fluids and their essential characteristic is able to reversibly change from free-flowing, linear viscous liquids to semisolids having controllable yield strength in milliseconds when exposed to a magnetic field. For this reason, it is extensively used as active devices for structural vibration control. The concept of structural vibration control is to absorb vibration energy of the structure by introducing auxiliary devices. Magnetorheological fluid dampers are, one of such devices, whose damping characteristics, can be modified by varying an applied current.

This paper presents the design, fabrication, testing and characterization of MR damper, mainly consists of pistion and cylinder. The copper wire is wound around the piston the fins in to generate a magnetic field. These models were developed based on the Bingham characteristics and the numerical analysis is carried out to estimate the damping coefficient and damping force under oscillatory motion. An experimental set up also established to characterize the damper and results are compared. Results demonstrate that MR damper was good semi-active candidate for vibration control.

Design and Development of Wing and Fin Deployment and actuation Mechanisms for a Tube Launched High Subsonic Vehicle

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The two basic inversions of four bar Mechanisms are designed and developed for an UAV wing and fins deployment. Initially, the wing is kept fully inside the fuselage by keeping the wing span in line with the fuselage axis. The wing has to be rotated 90° within 0.5 s such that its span is perpendicular to the fuselage axis and then it is locked to the fuselage. The basic concept of slider crank mechanism is used for the deployment mechanism. The wing being the rotating member is considered to be the crank and a pyrojack is used for applying the force required to operate the mechanism. In this paper, the mechanism is verified theoretically by both Kinematic and dynamic analysis to validate it against the time and space constraints. The wing is assumed to rotate with constant angular acceleration for the first quarter of the total deployment time and similarly it is decelerating for the last quarter of the total time, and the remaining being of zero acceleration, thus resulting in trapezoidal profile for the angular velocity. Finally, the mechanism is validated experimentally. The purpose of the air vehicle fins is to provide directional control and aerodynamic stability during flight. Prior to launch, the fins will be typically folded in a stored position about the missile body to conserve the storage space and to minimize handing and launch problems. The fins have to be rotated from the folded position to deployed position immediately after the air vehicle comes out from the canister or launch tube. It is particularly important that the fins have to be deployed quickly without damaging vehicle components and remain in the deployed position to provide stable flight characteristics to the air vehicle.

Perspectives and Challenges in Futuristic Aerospace Mechanisms

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Design, development, qualification testing and deployment of Aerospace and related mechanisms is a challenging activity in the area of satellite launch vehicles, missiles, weaponry, satellites, aircrafts and robotics. Over the years the country has made rapid strides and achieved commendable successes in the filed of aerospace mechanisms. Our country has wide spectrum of engineers and scientists well expertised in design, development, testing assembly and integration of such mission critical mechanisms.

Present paper deals with various mechanisms which were designed, qualified and deployed with the expertise gained over a period of time in the field of Aerospace Mechanisms and gives an insight into the futuristic challenges.

Simulation on Explosive Dynamics using Shaped Charges for Stage Separation Application

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Flexible Linear Shaped Charges (FLSC) are widely used in staging and destruction of missiles and launch vehicles. FLSC works basically on the "Munroe effect" principle i.e. total cut by explosive in target material comprises of jet penetration and shock severance.

FLSCs are tested on flat plates in simulated flight configuration with explosive encased in between target plate and backup segments. With EN-24 backup segments, target plate is being cut at more than two locations. Whereas, single cut is observed with aluminum alloy backup segments.

This paper deals with the simulation study on Explosive dynamics in FLSC by using 2D model of AUTODYN software. The simulations could predict well the performance of FLSC in the above configurations. It is observed that with EN-24 backup segments, the target plate severance is mostly due to shock wave, whereas with aluminum alloy, severance is by jet penetration. Severance is explained by considering the strength of the two materials En-24 and Aluminum alloy.

Usage of AUTODYN software reduces the product development cost and time. It is successfully used in our programs during the development to deployment phase of our systems.

Deployable Nozzles for ISRO Launch Vehicle

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The never-ending quest in chemical propulsion is to maximize the performance index of specific impulse. This depends on how efficiently combustion takes place and how effectively the hot gases are expanded. Higher nozzle area ratio means higher C and in turn means higher Isp. Very high exit nozzle diameters and corresponding nozzle height pose problems for the engine envelop and overall engine height and consequently the launch vehicles length. The emerging concept is to adopt a Carbon-Carbon deployable nozzle extension wherein the entire nozzle extension can be segmented and the segments can be stowed to reduce the height. During flight, these segments can be joined to form the full nozzle to yield the desired expansion efficiency by means of high precision mechanisms applicable to Aerospace environments.

Such an Aerospace mechanism for launch vehicle upper stage engine has been successfully developed for delta III launch vehicle. The technological challenges in the field of fine techniques has been well addressed in this paper.

DM/P/41

Design of an Inertial Grade MEMS Accelerometer for Space Applications

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Inertial grade accelerometers find several space and military applications right from being a part of inertial navigation unit to outer space exploration programs. Accelerometers with micro-g resolution in a band of d c to 100 Hz and low bias and scale factor drift are required to meet the specifications of such missions. Moreover, the reliability aspects of these sensors are very critical since they are subjected to harsh conditions and environments during launch time and the mission period. Silicon is an established material for such environments. Bulk micromchined Silicon accelerometers employing a capacitive sensing and force feedback electronics is proved to be the best approach to realize inertial grade accelerometers. This paper presents the design of a capacitive sensing accelerometer with interdigited comb construction and a split electrode configuration to realize an inertial grade performance. The sensor is proposed to be fabricated by high aspect ratio etching of silicon using Bosch process and will be employing Silicon on Glass architecture.

The structure designed is a silicon seismic mass supported by four beams bonded to a glass wafer at its bottom and is compliant to deflect in-plane. The beams are folded to increase its effective length and thereby the mechanical sensitivity. Combs are spread out of its vertical planes on both sides and are designed to be inter-digited with a set of fixed combs anchored on the glass wafer. The whole array of combs is distributed into four sets enabling an effective differential sensing mechanism. We have proposed a force feedback electronics for the measurement of acceleration which is a must to achieve the targeted specifications. The present arrangement of electrodes will help the electrostatic actuation of the structure. Our initial design intended to have two sets of electrodes on either sides of the seismic mass for differential sensing and one of the pairs of electrodes to be energized at a time for force feedback depending upon the direction of acceleration. But the simulation results showed that perfect linear displacement is not achieved on electrostatic actuation of structure. A novel approach of splitting the electrodes has solved this problem. Moreover, it offers an additional advantage of introducing a more effective built-in self-test feature to check health of the structure before getting integrated to read-out electronics. Stoppers are provided to protect the structure from large unexpected mechanical shock acting on the system while powered off.

The details of the structure, results of the mechanical and electro-mechanical analyses, and highlights of proposed fabrication flow using Deep Reactive Ion Etching (DRIE) will be presented in the paper.

DM/P/42

ASW Airborne Winch System: Design Challenges

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Emerging trends in Anti-Submarine Warfare (ASW) shows the extensive use of Helicopter based dunking systems. The deployment mechanism used for th raising / lowering of a dunk body is an airborne winch. In this paper, the main design challenges involved in the development of an ASW airborne winch are addressed. The design of an airborne winch is an iterative process involving consideration of various operational and platform parameters. The critical subsystems and the various criteria for component selection are also highlighted. The design process of an ASW airborne winch starting from the input design parameters, their effect on various subsystems and the iterative process are brought out in this paper. The performance evaluation of the winch and the qualification criteria involved in the airworthiness certification are also discussed. The capability of winch design software in the reduction of total design time cycle is revealed.

Design and development of Passive Stabilization System for Antenna on a Floating Platform

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This paper explains the design and development aspects of a Passive (not powered by external means) Stabilization system for the Antennae mounted on a floating platform. The purpose of the Stabilization system is to keep the angular orientation of the antennae within +/-5° from its ideal(horizontal) position in pitch and roll planes, when the antennae are mounted on a 10m height mast fixed on a marine float. The float upon which the antennae are mounted is under constant disturbance by the sea waves and may tilt up to +/-30° with cycle time ranging from 10 to 20 seconds. Due to this continuous disturbance offered by the sea environment, there may be the possibility of communication failure between the marine float and the other communicating ship.

In order to cater for the above problem, a stabilization system has been developed which is capable to keep the angular orientation of the antennae within the limits (+/-5° ,from its ideal position), to avoid any communication link failure. The stabilization system is fully mechanical and utilizes the force of gravity.

ABSTRACTS

Ejection Development & Recovery Systems

Wing and Fin Deployment Mechanisms for UAV

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This paper presents Wing and Fin deployment mechanisms for a UAV. Immediately after the UAV launch, wing and fins have to be deployed. It is required to deploy the wing and fins within specified time for smooth UAV launch.

Based on space constraint, ease of assembly and operation, single wing deployment mechanism has been selected. The loads on the wing and torque required to unfold the wing during deployment have been calculated. Wing deployment tests have been carried out and time required for deployment has been estimated. The test results are presented.

Internal spring and external spring options for fin deployment mechanism have been considered for the UAV. The torque required to deploy the fin has been calculated. External mounted torsional spring mechanism has been selected due to its ease operation and design details are discussed. Fin deployment tests have been carried out, time required for deployment estimated and test results are presented.

Design, Analysis and Development of Wrap Around Fin (WAF) Stabilization Mechanism for Artillery Rockets

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Artillery rockets are generally fired from Multi Barrel Rocket Launcher (MBRS). Free flight rockets fired through launch tubes have got wrap around fin type configuration to enhance the packing density for a given volume in a launch vehicle. The fins get deployed as soon as the rocket comes out of the launcher. In order to minimize the effect of thrust mis-alignment and dispersion at target end, fins are provided with cant angle. These canted fins generate the required spin to minimize the effect of thrust mis-alignment. A mechanism that enables folding during assembly and un-wrapping and positive locking of fins during deployment using a combination of torsion and compression springs either with submerged configuration below a ballistic/aerodynamic shroud or projected on the external surface and stabilizer mounting is generally employed. This paper brings out various WAF configuration stabilization mechanisms designed and developed successfully for artillery rockets. Merits and demerits of various configurations are discussed. Theoretical approach for estimation of cant angle and spin are presented in this paper. This is followed by presentation of results of detailed stress analysis of mechanism under aerodynamic load and inertia loads using finite element analysis software. Finally, simulation studies carried out on centrifuge machine and shock test machine to evaluate the effectiveness of the mechanism are also presented. This mechanism has been successfully flight tested.

In Flight Deployment Mechanism for Submunitions

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A 200 kg Penetration cum Blast type of warhead for a surface to surface missile is envisaged to neutralise hard concrete targets spread over an area of approx. 100 m x 100 m area. The missile is expected to reach the target area with a velocity of 400 m/s in a near vertical condition. The space constraints for warhead are taken as dia 400 mm and length 1000 mm. A conceptual design of the warhead is presented in this paper.

In the conceptual design, the warhead consists of qty. 6 nos. of submunitions which are held by a holding structure which in turn consists of plates and tie rods. Each submunition (having mass of 27 kg), is assembled inside a launch tube which in turn is integrated with the holding structure. The holding structure is en-housed in the missile nose cone with the help of bulk heads. To pack maximum number of submunitions, submunitions are initially kept at a stack angle of 0 deg with in the volume available in the nose cone. The nose cone is cut open before the submunitions are released from missile. Before ejection from the missile, the submunitions need to be unfolded to avoid collision with mother carrier. A mechanism is worked out by which the submunitions are rotated against aerodynamic forces to attain angle of 8 deg., before they are released. To maintain overall structural mass to a minimum level, it is proposed to eject the submunitions simultaneously. The submunitions are ejected at a predetermined altitude for effective dispersion in target area.

In this mechanism, the submunitions are pivoted to the top plate. A pyro cartridge, housed centrally is operated to generate the required force. The hot gases act on a wedge, on which six rollers connected to each submunition rest through a bracket. The linear motion of the wedge is thus converted to rotary motion of the submunition. All the submunitions are rotated to desired angle simultaneously. When the desired angle is achieved, the submunition is locked in that position. The paper presents the methodology, kinematics and force analysis of the design in detail.

ED/O/03

Design of Wing Opening Mechanism for an Aircraft Bomb

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Air warfare is one of the most effective means of attack to defeat deep enemy targets. With the advancement in surveillance and missile technology, the air-attack has to be carried out from a range far enough, not to expose the attacking aircraft to the enemy defence systems.

Among the various methods of range extension of a bomb, mounting the bomb with lift surfaces is considered to be a simple option. A glide-kit comprises of aerodynamic lifting surfaces (wing) kept in folded condition in a casing during carriage and opened by means of a mechanism after release from the aircraft. This mechanism should open the wings after the bomb is released and is clear of the pylons and the fuselage of the aircraft. Both the wings should be opened simultaneously through the same angle in fraction of a second.

This paper deals with choice of a spring motor actuated mechanism for Glide-Kit. The mechanism is a four bar Slider-Crank mechanism. Initially, the slider is actuated using a spring so that the mechanism can open up the wings quickly to an angle of 60°. At this stage the slider is actuated using a motor through a threaded shaft-nut arrangement to control the sweep angle of Wings. The wings are connected to each other through gear sectors which will ensure that at all times both the wings are opened to the same angle.

The mechanism is synthesised by studying the effect of various variables on the load on the rocker arm. The variables chosen for study are the location of the slider with respect to wing pivot, the length of the rocker arm and the starting angle of the crank with length of the crank kept constant. The study revealed that the value of the starting angle of the crank should be as small as possible for the overall load on the rocker arm to be the least. The value of the length of the rocker arm should be as high as possible and the location of the slider with respect to wing pivot should be as close to it as possible for the overall load on the rocker arm to be the least.

Then the kinematical analysis of the mechanism was done by deriving the equations of motion by the Lagrangian method using MAPLE symbolic software. These equations of motion are then solved by numerical integration using MATLAB software. This process was done in two steps, one, when the spring force is acting and; two, when the motor torque is only acting. The response of the system is then studied and it is found to be suitable for a Glide-Kit.

ED/O/04

Static and Dynamic Analysis of Reflector Deployment Mechanism

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A typical communication satellite is configured to have two antenna reflectors, mounted on the East and West sides of the spacecraft. These reflectors are stowed during launch and deployed in orbit. The holddown/release mechanism holds the reflectors during launch and releases the same on-orbit. With the release in the orbit, the deployment mechanism ensures rotation of the reflector about its hinge axis and latching-up in the intended final orientation. The deployment mechanism for each reflector consists of two hinge assemblies and two hold-down assemblies to clamp the reflector on the East/West faces of the S/C. The mechanism is subjected to dynamic loads during launch and latch-up-shock loads at the end of the deployment. The launch loads are shared by hold down and hinge assemblies and latch-up shock loads are withstood by hinge assemblies.

A mathematical model of the reflector deployment mechanism has been developed. using MSc PATRAN. This includes the finite element modelling of different constituents like hinge brackets, holddown assembly, locking linkage etc., These have been appropriately positioned on a spacecraft model, simulating the stowed configuration of the reflector on spacecraft. The finite element model of the reflector has been integrated with this to obtain the full mathematical model of the system. All the individual constituents have been connected together with the help of rigid beam elements. The requisite boundary conditions at hinges and holddown assemblies have been simulated.

Detailed analyses activities have been carried out using the above mathematical model of the system. The software NASTRAN has been used for the analysis. Initially, normal mode analyses on each of the subassemblies have been carried out to validate the corresponding mathematical model. Each of these subassemblies has been systematically integrated and at each stage, the normal mode analysis has been carried out to validate their mathematical assembly. The normal mode analysis on the full assembly provides useful design inputs like the natural frequencies, mode shapes, participating masses etc., Static analyses have been carried out on the sowed assembly, simulating the launch acceleration loads.

A deployment dynamics study has been carried out using the software ADAMS. The hardware characteristics of different constituents have been simulated (Figure-1). The results from the deployment dynamics study include determination of deployment time and latch-up velocity for on-orbit and simulated ground deployment conditions. This also provides inputs for deployment shock analysis.

Further, the finite element idealization of deployed configuration has been simulated.

ED/O/05

A normal mode analysis has been carried out to determine the deployed natural frequency of the system. Static analysis has also been carried out on the deployed assembly, simulating the latch-up shock loads. This analysis provided design inputs like reaction forces and moments at different interfaces. Latch-up shock analysis has been carried out to estimate the latch-up moment loads on the hinge line.

Subsequently, reduced mathematical models for hinge and holddown assemblies have been developed using Creig-Bampton technique. These are integrated with the finite element model of the reflector and the above activities have been repeated. A good match of results between reduced model results and full model results have been observed.

The paper presents details of the above analysis activities.

Design and Indigenous Development of a Snap-off Mechanism for Lauch Vehicle Cooling System

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Avionic system comprises Electronic packages, electrical harness, electrical connectors, mechanical mounting structures, electrically operated sensors, antennas etc. Such avionic systems are housed at many stage sub-assemblies in the launch vehicle and are the main tool to carry out various system functional checks and vehicle check out operations both on ground and in flight. When the avionic system is switched on, significant quantity of heat is generated by the packages as they perform functions and is coupled with solar heating from external surroundings, since the launch vehicle is exposed to ambient environment during the launch pad operations. In order to have normal functioning of the avionic system which is essential for normal performance of the specified temeperature limits till the end of the mission. To achieve this mission requirement, continuous supply of recommended quantity of conditioned cool air from Umbilical tower (UT) to avionic sub assemblies is carried out through a well designed cooling system during the various phases of electrical checks and functional checks till lift off from launch pad. The function of the cooling system ceases at launch vehicle lift off and the vehicle has to be disconnected from umbilical tower.

The cooling system comprises a ground half connector, a vehicle half connector, a set of flexiblke cooling hoses that connect both halves and a set of retraction cables anchored to UT. The disconnection of the ground half of the system at launch vehicle lift-off is the prime requirement of the cooling system,. The ground half of the system has to be smoothly disconnected from the ongoing half is aided by a snap off mechanism which demands a specific pulling load. This mechanism is designed for a specific requirement at Second Launch Pad (SLP) where the vehicle stand off distance from UT is ~ 12 metres. The design has to ensure no pre-mature disconnection at launch pad due to wind loads and the inertial mass of the colling hoses. The mechanism involves a set of spherical balls locked in grooves and a selective spring to match the design pulling load. For Re-usable Launch Vehicle, in addition to smooth snapping of the ground half, the port of the vehicle half connector is to be shut after disconnection against entry.

This paper focuses on the design and development of a snap off mechanism used for GSLV flight at second launch pad (SLP). It covers design specifications, material specifications, functionally critical dimensions, mechanism involved, qualification plan and incorporation of a shutter in the vehicle half to be operated after ground half disconnection.

ED/O/06

Hold Down, Release and Steering Mechanism for Observatory Payloads

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Many observatory satellite payloads, consisting of optical telescopes or cameras with associated electronic packaged mounted nearby, need to be steered about a satellite axis everyday for their observation orientation requirements. These payloads need to be clamped to spacecraft structure during launch and are to be freed for steering in orbit. The clamping/hold-down used should clear and provide for unobstructed steering of the released payload up on its release. A typical steering mechanism discussed in this paper has to steer the payload platform from 0 to 180^0 and back, covering one cycle in a year. The hold down, release and steering mechanism was configured to structurally support this payload for launch, release the payload after launch and to steer the payload optics axis to the required orientation.

The whole payload is mounted on a separate structural deck. The drive shaft of steering motor module in the centre is attached to payload deck through a pair of diaphragms and the drive module is fixed on to the structural deck of the spacecraft. The deck is held down to the spacecraft deck at four points by bolts and is released in the orbit. The hold-downs support the payload/payload deck for launch loads. This ensures that the launch loads are not transmitted to the bearings of the drive module. In the orbit, the payload deck is released and is pushed up by 3 mm to clear the hold down support points and to facilitate free rotation of the payload deck using two corrugated preloaded diaphragms. In the held down condition, these diaphragms are deflected by 3.5mm to have pops up preload.

Corrugated diaphragms are chosen to have a near linearity in force-deflection characteristics and to limit the stresses induced. Iterative finite element analysis of the corrugated diaphragms has been carried out to estimate the number and pitch of corrugations to have near linear force deflection characteristic in out of plane direction. As the deflections involved are large (3.5 mm) compared to the thickness of the diaphragms, a non-linear static analysis has been carried out to establish the force-deflection behaviour of corrugated diaphragm using MSC.MARC.

Upon release, the payload deck is rotated to the required orientation within a range of 0^0 to 180^0 by a steering mechanism. The steering mechanism essentially consists of a stepper motor and a pancake type harmonic drive reducer. The motor is fixed on to the input shaft by duplex pair angular contact ball bearings. The motor drives the input shaft of harmonic drive in steps of 1°. One gear is fixed to the outer housing and the other to the rotating side (called dynamic gear), which carries the output flange and is supported on a four-point contact bearing. The gear reduction is 157:1. The output flange interfaces with the diaphragm assembly, which in turn is attached to payload deck.

ED/O/07

Release dynamics of the payload has been studied. The development of hold-down release mechanism, popup mechanism and steering mechanism have been completed.

Structural model testing of the mechanism has been carried out. The performance of the mechanisms is nominal and met all functional requirements.

The configuration and design details, results of analysis and testing are discussed in this paper.

Ultra Low shock Stage Separation Mechanisms for Long Range Missile Application

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Stage Separation Mechanism is very Mission critical operation in long range missile systems. However the current trend is to look for a very low shock system during stage separation transition. Since the explosive shock generated will lead to the failure of the mission.

In order to tackle such a situation an Ultra low shock stage separation mechanism has been designed, developed, qualified and deployed for a specific missile application successfully.

And the same concept is going to be implemented in long-range missile systems due to the low shock features incorporated in the mechanism. This paper deals with the journey of the mission critical mechanism from design to development.

Development of Parachute Deployment System for SRE Mission

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A parachute based deceleration system is used during the terminal phase of SRE mission to achieve the targeted impact velocity to the capsule suitable for safe water recovery. The parachutes in a canister are positioned at the aft end of the capsule. The outer end of the canister is covered with a lid plate that is mounted in line with the base of the capsule. The parachute system deployment is initiated by deploying a pilot parachute followed by a drogue parachute when the capsule reaches 5 km altitude. Subsequently on reaching 2 km altitude, the drogue chute is disconnected and a main chute is deployed. The extraction and deployment of drogue parachute in the presence of wake flow at the aft end of the capsule, has posed real challenge to the programme. The thermal protection systems provided to protect the parachutes from the base heating expected during reentry period, further complicated the issue. This paper presents an overall description of the design, development and qualification programme followed for parachute deployment system along with a brief account on the failures experienced and lessons learned.

ED/O/09

Development of A Pyrotechnically Actuated Drogue Gun for A Parachute Deployment System

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Parachute based systems are the most common mode of deceleration systems employed in aerospace systems operating in the atmospheric region. The system is configured as train of parachutes starting from a small chute deploying a larger one and so on and finally the main parachute is deployed to effect the intended deceleration. Normally when a series of parachutes are used, a deployment device Drogue Gun is used to stretch and deploy the first chute of the parachute train. This initiates the deceleration system leading to the deployment of the main chute and generation of necessary drag to the moving body. In cases where the chute deployment has to be very fast a mortar based parachute deployment system is employed wherein the deployment of the bigger chute can be achieved directly.

The paper presents the development efforts on a pyrotechnically actuated Drogue Gun for the deployment of a parachute. The Drogue Gun comprises of a slug mass, which is projected out with a velocity capable of stretching and deploying the first chute pilot chute of the parachute train. Further, the slug mass velocity should not be too high so that the slug mass gets detached from the pilot chute prior to its full stretching. The function of the Drogue Gun becomes more complex due to the fact that factors external to the device influence the functional output of the device. Firstly, the parachute will have to be stretched and deployed across a highly dynamic environment and secondly, the capability of the slug mass to stretch and deploy the parachute will depend on the resistances from the parachute as well as other system elements. The Drogue Gun specification finalization like the mass of slug mass and its velocity becomes difficult and is highly dependent on the sub system configuration including the parachute configuration. The design approach, methodology adopted for finalizing the specification, testing for proper functional evaluation and simulated system level tests of a Drogue Gun used in the deceleration system of a re-entry module is highlighted in this paper.

Design of Aeroconical Parachute for Recovery of Lakshya Drone

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A two stages recovery parachute system for Lakshya (weighing 500 kg) an inter-services target aircraft has been designed to recover the flight vehicle after completion of its mission or on onboard emergency. Exiting Recovery Parachute System is a tri-conical canopy and being for one streaming only. A need was felt to improve the parachute for multi streaming application.

The suitability of state-of-the-art aero-conical canopy designed which gives higher stability in recovery of DRONE. The model study of similar kind of parachute has been conducted at IIT Kanpur and coefficient of drag is found as 0.635. Material of the parachute has selected such way that it can be used for minimum 10 streaming, however re usability has to be established by drop/RTRS trials.

ED/O/11

Design & Development of Parachute system for Remotely Delivered Munitions

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In view of fast changing battle field scenario, state of art technologies and tactical requirements, the RDMs are gaining utmost importance. ADRDE received a OR for design & development of a suitable stabilization system for Remotely Delivered Mines (RDM). The minimum release altitude was 500 m and max. packing space was only 195 cc. the Parachute must provide stability after 0.4 sec release from warhead.

Initially two prototypes were made from two types of fabric Nylon (37 gm & 52 gm). Both type of parachute was found to be pack able in the required packing space. These prototype were tested in RTRS trial at 345 m/s which is equivalent to 360 m/s at 500 m AGL. Prototype made up of 52 gm found successful in these trials, subsequently this parachute found satisfactory in dynamic trials. But in technical trials held at Pokharan, two parachutes were found damaged. Failure analysis of these parachute were carried out at ADRDE and parachute were strength as per recommendation of the board. This strengthen parachute performed satisfactory in 35 dynamic trials.

This paper discusses the design requirement, design approach, brief description of parachute system, operation, delay for deployment of parachute, selection of suitable shape of parachute, snatch force and opening shock force calculation, design validation using RTRS trials & module drop trial to establish rate of descent.

Design and Development of A Pyrocutter Mechanism and Parachute Release Mechanism for Vertical Erection of Minelets during Deployment of Remotely Delivered Anti Tank Minelets from MLRS

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A mechanism has been successfully designed and developed for vertical erection of submunitions which is a must for shape charged directional A/T minelets for penetration in the belly of the Tank. This mechanism is successfully implemented in the RDMS A/T minelets developed by Armament R&D Establishment, Pune. This Mechanism is very simple in operation and versatile for use in submunitions developed for RDMS and PINAKA warhead. It consists of pyrocutter, leg springs and Parachute release mechanism assembly. Pyrocutter is responsible for opening of the leg springs for vertical erection of minelets after landing on the target area and Parachute Release Mechanism (PRM) functioning. After arming of minelets during ejection and deployment of parachute assembly, battery gets connected to firing circuit. A delay of two minutes is given after arming of minelet at a height of 500-700m to the pyrocutter. Pyrocutter gets an electronic pulse after two minutes. Pyrocutter initiates and cuts the copper wire which holds the spring loaded folded legs. The mine gets erected and PRM gets thrown off. Design of pyrocutter, erection mechanism design and Parachute Release Mechanism (PRM) design etc. are discussed in this paper. Tests and validation trials conducted under various environmental conditions like on loose sand, under water & marshy land, are presented in this paper. The deployment sequence of minelets after pyrocutter functioning are captured using high speed photography, EOTS & CCD are presented in this paper.The ultimate performance evaluation was done in PFFR Pokhran ranges. More than 90% performance was achieved.

Design and Development of Ejection Mechanism for Submunition Child Store of Artillery Rocket Fired from MLRS

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A mechanism has been successfully designed and developed for remote delivery of Anti tank minelets.This development is a significant stage in the technology developed for the deployment of remotely delivered minelets as deterrent to the advancement of enemy's armoured coloum. This mechanism is used for smooth ejection of child store from the warhead, at very high velocity and altitude in flight. This mechanism ensures very controlled and safe base ejection of submunitions at a velocity of around 250 m/s. Ejetion mechanism is successfully implemented in the Remotely Delivered Minelets System based on GRAD 122mm rocket launched by BM-21 launcher. The paper brings out the sequence of functioning of mechanism, its design requirements, design approach used and test and evaluations carried out to achieve the smooth functioning with typical simulation study, experimental results, set up to optimize the charge mass, velocity measurement of minelet ejection, measurement of pressure time profile during initiation of charge mass etc. As many as 40 A/T rockets can be fired in 20 seconds up to 15 km range to cover an area of 1600mx250m target area. More than 400 rockets are so far fired with this child store ejection mechanism successfully.

ED/P/14

Variable Porosity Conical Ribbon Brake Parachute for LCA

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Brake parachute is used as a deceleration system for aircraft while landing. Aircraft requires high aerodynamic stable and low opening force parachute that does not disturb its inherent stability. LCA uses unicross parachute as brake parachute. Based on various problem observed with available unicross parachute, new conical ribbon parachute of variable porosity emerges out as a possible solution as brake parachute. Conical ribbon parachute of same drag area as of unicross parachute produces 5 10 % less opening shock force. Conical ribbon parachute is more stable as it is evenly porous through out the canopy. Conical parachute produces 10% higher drag than flat canopy parachutes. Conical ribbon parachute does not generate any rotation and do not interfere with aircraft's inherent stability. The variable porosity increases the drag coefficient as well as the stability. Conical ribbon parachute enables us to use rigging line length up to 1.5 times of nominal diameter, which in turns result in increased drag coefficient, larger projected diameter and high stability. Conical ribbon parachute are used successfully up to Mach 2 for rapid deceleration, which increases the operational speed range. Canopy with large rigging lines reduces the tendency of gradual canopy closing at supersonic speeds.

Conical ribbon parachute is already in use as brake parachute in Jaguar aircraft.

ED/P/15

Auxiliary Systems for Recovery of a Re-entry Module from Sea

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India proved its capability of bringing its satellite back to earth when Indian Space Research Organisation successfully conducted its first re-entry experiment, Space capsule Recovery Experiment, (SRE) in January 2007. The re-entered module made a safe touchdown in the Bay of bengal, about 200km off the coast of Chennai on 22nd January. Though it was the first attempt, within an hour, the joint team of ISRO and Indian Coast Guard could successfully recover the module from sea. This paper brings out the experiences gained while developing the various auxiliary systems required for the recovery of SRE module from sea. The auxiliary systems include pollution tester for testing presence of propellant traces around the capsule, set of inflatable buoys for enhancing buoyancy of capsule, set of inflatable buoys for enhancing buoyance of capsule and towing to the ship. The pollution tester has special chemicals in it and which will turn the colour of sea water sample depending upon the chemicals contained in the sample. This testing is done by lowering the device from a hovering helicopter and once the test is negative, clearance will be given for the recovery crew. Then a set of auxiliary floats are attached to the capsule to enhance the buoyance of capsule. A toroidal float will be put around the capsule for easiness of towing the capsule to ship. After towing the capsule to ship, a special net is used to lift the capsule to the ships deck. The capsule will be lowered into a special container, which is partly filled with water to dilute any trace of propellant oozing out from the capsule.

The auxiliary systems got modified based on the experiences gained from the numerous trials conducted in sea. All foreseeable conditions were simulated in the trials which resulted in a fast recovery of the capsule after splash down.

Concepts for Wing Opening Mechanism for an Aircraft Bomb

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It is gradually being universally accepted that the fighting qualities of a modern airborne weapon system not only depend on flying performance and on the efficiency of its avionic suite but also in at least equal measure on the sort of payload it carries. Airfields, important industrial complexes and installations are most heavily defended targets. Aircrafts attacking such targets normally find themselves confronted with a large number of highly effective air-defence emplacements. It is, therefore, important to use standoff weapons along with precision guidance system to attack enemy's installations without coming in the range of its air defence systems.

Some of the methods to increase the range of the bomb are: Attaching a rocket motor and attaching lifting surfaces to the bomb. Among the various methods of range extension of a bomb, mounting the bomb with lift surfaces is considered to be a simple option. A glide-kit comprises of aerodynamic lifting surfaces (wing) kept in folded condition in a casing during carriage and opened by means of a mechanism after release from the aircraft.

This mechanism should open the wings after the bomb is released and is clear of the pylons and the fuselage of the aircraft. Both the wings should be opened simultaneously through the same angle as quickly as possible. The mechanism should be light weight, highly reliable, easy to manufacture and economical. Various mechanisms considered to open the wings are:

1. Vane operated and Pulley driven/Gear driven mechanism
2. Motor operated and Gear driven
3. Power Cartridge Operated and Piston/Link mechanism
4. Helical Spring Operated and Piston/Link mechanism
5. Leaf Spring operated mechanism

This paper describes the construction and working of all the above mentioned mechanisms and their various advantages and disadvantages.

ED/P/17

Aircraft Ejection Seats - Advanced Concepts

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In early days, ejection seats of fighter aircraft used catapult or compressed air and simple mechanical springs. All these systems were dangerous resulting damage to the vertebra column of the pilot. When the speed of aircraft is low, the pilot was to jump out with his parachute. But as the speed increases jumping out from the aircraft with the parachute were merely a matter of luck and not guarantee for survival. The escape of the aircrew from high-speed aircraft was serious drawback experienced during World War-II. The jumping out of aeronaut from high-speed aircraft was extremely difficult because of high wind pressure, turbulence around the exposed cockpit and accelerations during the maneuvers of aircraft. In addition, immediately after the aircraft left the seat, there was a risk of collision with the tail or any other part of the aircraft structure. Therefore in several cases, fatal injury to the pilot during the ejection takes place. All these were proved hazardous, which mostly resulted in compression structure of vertebra column of the pilot. Therefore only explosive assisted ejection of aircraft seat is the latest and most effective. In all modern fighter aircraft the whole sequence of ejection is fully automatic once the pilot pulls the manual operated lever till he landed safely. The latest ballistic seats are equipped with telescopic ejection gun system, rocket pack, harness system, face mask, emergency oxygen system, survival pack, canopy severance system and drogue parachute system. The use of an ejection seat is always a last resort when an aircraft is damaged and the pilot has lost control. However, saving the lives of pilot is a higher priority than saving planes, and sometimes an ejection is required. Emergency escape from aircraft has been utmost importance to Indian Air Force since its inception. Regulations and policies to ensure the safety and survival of crew - members have been major thrust of entire safety program in Air Force. An attempt has been made in this paper to discuss the advanced ejection seats with their improved performance capabilities to the goal of improve the survivability of pilot during emergency escape.

Aircraft Emergency Ejection Seats, Canopy Jettisoning Systems and Power cartridges : Current Status and Future Trends

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The most important and valuable asset of Indian air force is its fighter aircraft / trainer aircraft. Considering the enormous cost, time and efforts for building up a well conversant air force pilot with the highest efficiency, it is also equally important and extremely precious are the lives of pilot and their services too. All the Combat and Trainer aircraft are provided with air crew escape aid systems for safe rescue of the pilot throughout the flight envelope. Only means of escape for the fighter pilot is the ejection seat and successful operation of the ejection system whenever aircraft emergency occurs at any phase of flight envelope. The highly reliable ejection seat systems are universally provided with different types of power cartridges in the sub-systems of the aircraft seat.

For the past three decades Armament Research & Development Establishment (ARDE), PUNE had been the main development and production agency for various power cartridges needed by Indian Air Force and Navy. These cartridges after successful development are produced at Air Pilot Plant at ARDE till Transfer of Technology to Ordnance Factory is established. Thus, the Air Pilot plant at ARDE established in 1970s to meet the service requirement is a strategically important asset of DRDO contributing to the operational readiness of Combat Aircraft and Helicopters of Indian Air Force and Navy.

Till date ARDE has developed about 67 different types of cartridges right from Canberra to Mirage 2000 and MiG 29 aircraft. Out of these, production of 26 types, whose requirement is more, has since been established at various Ordnance factories like OFK, AFK and OFDR. Out of the remaining cartridges, which are currently in service, about 30 types are still being produced at Air Pilot Plant only.

The strategic importance and mission critical requirement of these cartridges are best understood from the fact that the combat and trainer aircraft could have to be grounded in the absence of these life saving devices.

This paper brings out evolutions occurred in the ejection system, canopy jettisoning systems and their sequence of operations, during ejection with the use of high energy sources i.e. power cartridges. It also deals with various types of ejection seats and related power cartridges in the various fighter aircraft.

ED/P/19

Dynamics of Payload Release Mechanism

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India's first multi-wavelength space platform for astronomical study of cosmic sources consists of a host of scientific payloads. One of these payloads has a requirement of a steering mechanism to scan the whole sky in search of x-ray sources that become active. This Payload is mounted on a platform, which is interfaced to the steering mechanism which steers the latter to enable the complete coverage of the sky.

The steering mechanism used is capable of rotating the platform clockwise and anti-clockwise continuously by ±180°. The steering mechanism consists of a stepper motor, a compact harmonic gear drive and an encoder, which provides the angular position of the platform. To offset the launch loads from the bearings, the payload platform is floated on a pair of diaphragms, which isolates the launch loads to hard points on the structure through the hold down blocks. On release of the hold down mechanism the diaphragms popup and the payload platform separates from the hold down blocks and the payload is free to rotate.

The diaphragms are required to have a near linear and low stiffness in axial direction with high radial stiffness to ensure minimum loads on the bearings of the steering mechanism. In order to arrive at the same, detail analysis of various parameters of the two diaphragms were carried out and the top & bottom diaphragms were selected so as to have a near equal stiffness characteristic. The stress analysis of the same has been carried out using MSC PATRAN, NASTRAN and MARC and the same is reported in the paper. The paper also discusses about the dynamics of the payload on release.

ABSTRACTS

Safety
&
Reliability

Sub System Hazard Analysis (SSHA) and Failure Mode Effects & Criticality Analysis (FMECA) of In-flight Egress System of CSS for Trainer Aircraft.

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The trainer Aircraft is two seater aircraft having two cockpits, one for the trainee & other for the trainer. The Canopy Severance System (CSS) for trainer aircraft has been designed and developed for Integration into Ejection system for Pilots rescue in the quickest possible time during emergencies. The advance Technology of this type is being developed for the first time in the country. Canopy Severance System is being adapted in through canopy ejection. The pilots along with seats will be ejected out through the canopy bubble, which will be weakened by making a through cut by explosive cord at the center line of canopy. Canopy Severance System is configured in two independent Systems viz. In-flight Egress System (IES) & Ground Egress system (GES). Since high energy materials are used for functioning of system & the systems are to be used in emergencies; the correct operation of the system becomes critical. The paper brings out an attempt made to verify by using Sub System Hazard Analysis and Failure Mode Effects & Criticality Analysis that adequate safety & reliability measures are built in the design of In-flight Egress System, so that there are no operational hazards encountered during Life period of the system. The analysis brings out that the design of the sub system is satisfactory.

SR/O/01

Role of Canopy Severance System (CSS) In Seat Ejection for Military Aircraft

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Modern military aircrafts are developed with capability of flying at very high speeds and maneuverability to have an edge over enemy's flight during war or conflicts. As with this attempt to advance in flying speed an element of risk is also involved and hence necessary control measures during an emergency of flights are inevitable. To provide a form of safety/insurance in the event of aircraft accident due to various unavoidable circumstances, a means of escape from disabled aircraft has been provided by the way of Seat Ejection System. This seat ejection system provides a quicker action for pilot to escape from the endangered aircraft, so that the parachute in the system will have sufficient time to open-up and save the pilot before the aircraft crashes on the ground. Hence the time is very essence in the seat ejection sequence of an endangered aircraft. However before the seat is ejecting out of the endangered aircraft, the canopy has to be either removed from the position, or preweakened so that when pilot ejects out he does not encounter any incapacitating injury. However to remove the canopy from the path of the ejection seat, takes time and this time must be added to the total time taken for safe escape. In the many of the present day's aircrafts fitted with a sequence ejection system, it takes time delay of upto 1400 milliseconds for canopy removal mechanisms. But keeping the view of the present aircraft speeds and altitudes with which it is flying, pilot will be having, hardly few hundred milliseconds only, before the aircraft crashes in to the ground. Therefore need has been felt to design superior system for canopy removal for the safe escape of pilots.

This paper deals with the role of Canopy Severance System (CSS), which will reduce the time taken for escape out from the endangered aircraft to a minimum possible (20 ms) so that the reliability of the seat ejection will be higher and the pilot life will be saved without any incapacitating injury.

SR/O/02

Certification Approach for R-73E Air-to-air Missile Integration and Firing from LCA (Tejas)

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LCA is being designed as light weight, multirole and supersonic aircraft to meet the stringent & futuristic requirements of Indian Air force as its frontline, multimission single seat tactical fighter aircraft. LCA is being designed & developed by ADA in association with HAL and having more than 100 work centers in within the country, which are directly & indirectly contributing in LCA programme.

LCA the state-of-the art technology aircraft is being designed as a precision weapon launch platform to carry and deploy a vide range of weapons and stores with a quick turn round time. First mile stone of the LCA programme towards the weaponisation was to integrate Russian origin Missile R-73 (CCM), which is a standard fit on Sukhoi-30 aircraft.

Though the weaponization of LCA is highly complex and challenging task for the designers; it is equally challenging for the certification agency to evaluate the multidisciplinary requirements related to structures, armaments, aerodynamics, control law, electrical system and aircraft safety pertaining to R-73E integration and firing from LCA.

This paper brings out the approach/route followed for the airworthiness clearance of R-73E integration and firing from the state-of-art technology aircraft LCA. The certification approach covers integration of dummy R-73, training missile UR-73 and live missile R-73 for which various studies & analysis were undertaken such as mechanical load analysis, structural integrity, pylon design, CLAW analysis for carriage & firing, electrical design, release mechanism, failure analysis, analysis for effect of plume on structure & air-intake, flight instrumentation plan and overall safety of the aircraft. The route for airworthiness clearance of R-73 CCM firing has been established by the CEMILAC, which will be guiding tool for futuristic certification tasks for not only weaponization of LCA, but for other flying platform too.

SR/O/03

Role of Fault Tree Analysis in Risk Management of a Launch Vehicle

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Launch vehicle program always poses safety implications from lift off till orbital injection. Prior to the launching of a missile or space booster, an assessment of in-flight hazards posed to life and property is normally required. The hazards can be caused by the unplanned deviation of the stage / debris during flight resulting in instantaneous impact points (IIP), to fall over landmass / other facilities like oil wells etc. Risk is an expression of the probability of a mishap in terms of hazard probability and hazard severity. Analysis of risk implies identification of undesirable events, which could lead to materialisation of hazard, analysis of mechanisms leading to undesirable events, estimation of undesirable consequences and frequencies with which they could happen. This paper describes the role of Fault Tree Analysis (FTA) to arrive at a quantitative measure of hazard probability with the help of a case study where land impact is taken as a mishap. For a qualitative measure, hazard severity can be categorised as catastrophic, critical, marginal or negligible. To have a quantitative measure of hazard severity a measure based on the casualty area and density of the population of Landmass is adopted.

The results of risk analysis are used in the risk management allowing the comparison of estimated risk levels with those set as objectives in a particular activity (acceptable level), helping to set priorities for risk reduction strategies.

The resulting risk estimates also indicate location of high-risk areas and the overall level of the hazard and as a result, a revised trajectory or launch vehicle configuration may be required or high-risk locations may be partially or wholly evacuated.

Tuning MEMS-COTS for Space Applications

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MEMS are ideal for space applications due to its low mass, low power consumption, small volume and possible integration with control and sense electronics. Considering the challenges for new space mechanisms, the high reliability achieved with MEMS terrestrial applications, and part number reduction in MEMS enabled by high integration, it is conceivable that carefully designed MEMS could be more reliable than the conventional solution. But space qualified MEMS are not available as on today and reliability data on COTS (Commercially-Off-The- Shelf) is often unknown or unavailable. Thus, one of the key issues governing the acceptance of MEMS in space applications is the qualification of the devices in launch and space environments. These sensors are not specially designed and fabricated for use in space environment but which can be specialized with minimum technical efforts. These relate to operating principle, Microsystems, process modeling, circuitry design, process technology, and environment during application, packaging and manufacturing system. While the race for MEMS technological innovations is intensifying all over the world, development of MEMS technology in India is at its infant stage only.The goal and challenge of the present century is to take advantage of commercial MEMS-COTS (termed as MOTS) for space systems, while retaining sufficient reliability requirements to ensure mission success.

The most of silicon circuitry is sensitive to temperature, moisture, magnetic field, light and electromagnetic interference. Microscopic mechanical moving parts of MEMS have also their unique issues. Also, Packaging of MEMS is considerably more complex as it serve to protect from the environment, at the same time in contradiction, it enables interaction with the environment in order to measure or affect the desired physical or chemical parameters. Reliability depends on the mutual compatibility of the various parts of the MEMS with respect to the desired functionality and the choice of design and materials from the standpoint of long term repeatability and performance accuracy. Besides, reliability requirements varies from one application to another especially systems incorporating unique MEMS and reliability evaluation might be done by a methodology derived from the Physics of Failure approach. To determine the reliability of MEMS, one must understand the root cause of all relevant failure modes using a rigorous physics-based approach.

The growing acceptance of MEMS in safety critical application on earth as well as in consumer electronics is a testament to the high level of reliability that can be achieved in suitably designed and packaged MEMS. While the race for MEMS technological innovations is intensifying all over the world, development of MEMS technology in India is at its infant stage only. This paper addresses the current status of MEMS reliability and packaging technology and finally, identifies a need for a systematic approach to facilitate the insertion of this technology to high reliability space applications.

SR/P/05

Studies on an Automated Process for End Crimping of Explosive filled Tubes and its Merits Over Conventional Crimping

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Flexible linear Shaped Cords (FLSC) and Mild Detonating Cords (MDC) are used generally in stage auxiliary systems of launch vehicles such as stage ignition, separation and destruct systems and in air craft crew escape systems. The sheath materials used in these cords are lead, antimonial lead, aluminium, copper, silver etc. One of the important process steps is end crimping of the tubes before and after filling the high explosive charges like RDX, PETN, HMX and HNS.

Tube crimping is normally carried out manually. Space Ordnance Group of VSSC realized a custom built tube crimping machine with the help of an Indigenous industry. This automated machine is remotely operated and process standardization on end crimping was carried out for tubes of sizes varying from 6 mm to 23 mm in outer diameter and 50 mm to 700 mm in length.

This paper describes, in detail, the efforts taken for standardization of the automated end crimping process, the implication of this process automation on the performance of end products, improvements in safety aspects during the crimping process and quality aspects of the end products. A case study on two of our products namely Explosive Transfer Assembly (ETA) and square shaped mild detonating cord (square MDC) has been presented in this paper to compare the relative merits of automated end crimping over the manual one.

ABSTRACTS

Development Testing & Evalution

Design and Fabrication of Cross Corrugated Recuperator Fins for Micro Gas Turbine

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Micro Gas Turbine (MGT) is a new type of combustion turbine used for stationary energy generations which could produces 5-500kW of energy and can be located on sites with limited space. MGT offers many advantages over other technologies for small scale power generation, emission less hybrid vehicles, other advantages includes less maintenance and longer lifetimes because of a less number of moving parts with miniaturization technology, compact size, lighter weight, greater efficiency, lower emissions by introducing recuperator and quicker starting. MGT also offers opportunities to use verities of fuels.

National Aerospace Laboratories Bangalore is involved in development of 10kW micro gas turbines. The unit is comprised of a compressor, combustor, turbine, alternator, recuperator and generator. The compressed air flow through recuperator channel is mixed with fuel and burned under constant pressure in the combustion chamber. The resulting hot is allowed to expand through a turbine to perform effective work. The remaining waste energy is allowed to flow through the recuperator in the counter direction to the compressed air in alternate channels. The recuperator unit recovers some of the heat energy from turbine exhaust and transfers it to the incoming compressed air stream.

A recuperator is a mandatory for MGT to boost thermal efficiency of the unit. Recuperator is one of the largest cost components and it is labour intensive to fabricate. The thermal efficiency of the recuperator depends on fins geometry used. This paper describes in detail the design and fabrication of new fin geometry to increase the thermal efficiency of the MGT unit. This is achieved by increasing the surface area of the fin per unit volume by cross corrugated geometry. The recuperator fins for a micro gas turbine comprises the steps of stampings of spaced integral ribs using a metal sheet (SS 304) so as to define high pressure channel (Corrugated surface) and low pressure channel (Rib surface) which is used to integrate with annular recuperator. A special die sets are designed and fabricated to manufacture the corrugated fins for recuperator. The various fabrication technologies involved such as stamping process, spark EDM, wire EDM, laser welding and three-dimensional manufacturing techniques with more traditional micromechanical manufacturing processes are discussed in detail.

Design, Development And Testing of a Winch Mechanism For Parachute Assisted Soft Landing

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Air dropped Cargos and UAVs are landing with the help of parachute with a descent velocity of six to eight meters per second. The impact on ground can produce high deceleration forces (thirty to fifty times earth's gravitational acceleration) that are detrimental and cannot be tolerated by many equipments and materials.

An all weather rapidly re-deployable retraction winch mechanism has been designed and developed, which can neutralize the descent velocity of a parachute supported payload by pulling the payload up with respect to parachute just before touch down so that the touch down velocity is made nearer to zero resulting in soft landing with reduced deceleration levels.

The retraction winch mechanism consists of twin pneumatic cylinders with hollow lockable plungers, external winch assembly mounted in between the cylinders, pyro device for unlocking the plungers and ground proximity ultrasonic sensing initiator. Nylon braided Kevlar rope is anchored at one end and the other end is passing over the pulley blocks in the winch and taken out to the parachute. The payload is connected to the mechanism at the bottom.

A high-pressure charge is stored in the space between the cylinders and in the hollow pistons. When acoustic height sensor in the bottom of the payload determines the ground within 2 meters, on-board electronics fire a pyro device that unlocks the plungers putting the winch in motion. As the plungers travel the length of the cylinders, the pulley blocks of the winch are pulled apart retracting more than two meters of Kevlar rope in less than half a second.

The preliminary design of the pneumatic cylinder has been carried out for possible failure modes and FEM analysis is carried out to evaluate the performance against the design criteria.

3-D digital models are made to conceptualize the design, animation and for verification of the fabrication and assembly requirements.

Four prototypes of the cable retraction mechanism are fabricated and tested with features that are incrementally added for intermediate verification for limit pressurization of 150 bar. Laboratory tests are conducted to validate the design of pulley block system and to measure the terminal velocity of the winch.

A Quadra-pod type Para Simulator (test rig) with a height of 13.1 meters to accommodate an aircraft of 5.5 meters long and 8 meters wing-span for a load of 1000 kg has been designed, fabricated, erected and commissioned at ADE to simulate the para-descent velocity with suitable speed control to carry out drop tests. The design has been verified by FEM analysis for strength, stiffness and stability.

TE/O/02

The prototypes with integrated distance sensors are drop tested in the test rig and the terminal retraction velocity of the mechanism is measured using accelerometer and high-speed photography.

Pliable skids with swiveling supports have been developed to avoid toppling of the payload and landing on uneven ground. Samples have been fabricated and tested.

The concept of retraction for parachute assisted soft landing was successfully designed, fabricated and demonstrated by drop tests.

Development Testing of A Smart Actuator for Satellite Propulsion Feed System

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Smart actuators belong to a class of devices which utilize materials that are highly responsive and have the inherent capability to sense and react according to changes in environment. The common characteristic of all smart materials is the ability to react mechanically to external stimuli. Magnetostrictive materials respond to magnetic fields. The recent material advances in the area of magnetostriction, mainly Giant magnetostrictive materials (GMM) with large strain capabilities have created new design options. It has evoked considerable interest for these actuators in the field of air and space, particularly for functions offering 1) High mechanical energy density 2) Low power consumption 3) Vibration resistance 4) High resolution and 5) Fast response.

One of the most promising areas for application of this technology is for satellite propulsion using electric thrusters. Electric thrusters operate at a very high specific impulse and at very low thrust and hence they require a very low propellant gas flow rate and a tiny valve stroke. Accurate flow control of propellant is an important parameter in the design of the propellant feed system. The existing feed system provides two stage propellant control with a Pressure control module for presure reduction and a flow control module for flow control. An alternate approach being proposed would be to develop a proportional flow control valve with a large pressure turn down ratio. This simplifies the two stage propellant feed system to a single stage propellant feed system. Also it would provide reduced sensitivity to ground handling and contaminaion.

A proportional solenoid flow control valve using magnetostrictive technology is conceived. It essentially consists of a magnetostrictive actuator coupled to a valving unit. Development of the actuator would be a major milestone in the realization of the proportional solenoid valve. With this background a proto smart actuator is realized and subjected to testing. Optimisation of the envelope size, simultaneously achieving the desired stroke and power requirement is the design challenge. The magnetostrictive material used is an alloy of Terbium,Dyspsorium and Iron ($Tb_{0.3}Dy_{0.7}Fe_{1.9}$) The actuator consists of the magnetostrictive material surrounded by a solenoid coil . The material is prestresed and fixed at one end. It is free to elongate at the other end. On application of the magnetic field, the material deflects. The actuator stroke is computed for varying magnetic field and prestress and the material characteristics are plotted.

The present paper discusses the development and testing of the magnetostrictive actuator to be used as actuation element of the Proportional solenoid valve. The results are evaluated with respect to the critical requirements of stroke and power.

TE/O/03

Development Approach of Power Cartridge for Fighter Aircarft

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Ejection system in the aircraft is one of the most complicated systems having many sub systems. This system is used to rescue the pilot with in the shortest possible time from the disabled aircraft when emergency occurs. Power cartridges are installed in the aircraft cockpit, and used to operate these ejection systems. These cartridges are single-shot operating device and their performance can not be tested by Non Destructive technique. Hence, systems are required to design with utmost care to function with high reliability. The life of the pilot is precious therefore; their life must be protected against adverse circumstances. For this purpose necessary cartridges were successfully indigenized by ARDE, and are being supplied to users and many lives have been saved. Such airborne stores were developed on the basis of matching performance with that of imported store. The applications of these cartridges to jettison the seat with the pilot from the endangered aircraft, drop the weapons / empty fuel tanks to have positive separation from the parent aircraft, extinguish the fire in engine / aircraft in case of accident and cartridges like cutting the cable, giving distress signal. The performance evaluation of the cartridge is carried out in the closed vessel in the form pressure time curve. The area under this curve indicate the energy requirement of reliable operation various system in the aircraft. This paper brought out the details of various exhaustive trials conducted for the power cartridges for their stringent design to meet the users requirements. These air borne stores were subjected to life assessment trials to study performance degradation to assign storage life including the installed life.

TE/O/04

Development of Time Delay Cartridges for Firing Mechanism of Heavy Drop Platform System for AN-32 Aircraft

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In heavy dropping system of AN-32 Aircraft, Time Delay Cartridge (TDC) is used as subcomponent of firing mechanism assembly, which further becomes a part of automatic disengaging unit (ADU). The ADU is used to disengage / separate the parachute system from the cargo load dropped from aircraft platform system at the instant of its touch on the ground. Separation of parachute from the cargo load is mandatory to prevent it from being over turned or dragged by the wind force. 14 s and 20 s Time delay cartridges have been developed as per the maintenance manual of heavy dropping equipment to allow the heavy weight para dropping from below and above 500 m altitude respectively. 14 s TDC is developed by adopting insitu filling of delay tube while 20 s TDC has a compact coil of filled lead tube. The design may be tailored with flexible delay timings for heavy weight platform system of other aircraft too. Both the items have been introduced into services. The design features, methodology adopted for performance evaluation and QA aspects for pilot lot productionisation of both the TDCs are presented in the paper.

Process Technique Development for Flexible Linear Shaped Charges with Optimum Cutting Capability for Staging Application in Launch Vehicles.

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Flexible Linear Shaped Charge (FLSC) based separation systems are used for lower stage separation as well as strap on motor separation of launch vehicles where reliable cutting of the structure is highly demanded. Linear shaped charges are explosives encased in a seamless metal sheath designed to cut metal or other nonmetallic materials. They are manufactured in continuous lengths, shaped in the form of an inverted "V". When detonated, the V-shaped metal liner collapses forming a high velocity jet which impacts the target with pressures exceeding the target's yield strength and literally pushing the target material to either side of the path of the jet. For separation application, the FLSC is so designed that on the total cut of target, 50 % is achieved by jet penetration and the balance accounting for shock severance.

The cutting action can be made optimum by controlling the FLSC dimensions, configuration, explosive type, linear charge density and bulk density distribution across the section. Critical parameters such as mass of jet, length of jet, velocity of jet are to be made optimum so as to get efficient target penetration. Process technique for FLSC is very important in order to achieve uniform bulk density as it decides the maximum target cutting capability. An optimum designed cord will impart less disturbance to the ongoing vehicle.

FLSC based separation system selected for one of the staging application, consists of a hollow tubular structure with material 4.2 mm thick 17-4 PH steel heat treated to 40 HRC and having diameter 80 mm. Conventional FLSC with apex angle 90 ^{0} and caliber 19 g/m processed by rolling method have been used for initial trials. When the cord is tested, separation did not take place, but secondary cracks were observed to the adjoining structure. Further to realize reduced charge with improved performance, FLSC has been processed with a combination of rolling & swaging route and with reduced apex angle. Jet penetration tests on tapered plates and detonation velocity (VOD) measurements were done for both types of cords. Studies conducted showed increase in both jet penetration and VOD for swaged cords. Further trials were done for charge optimization. It is concluded that FLSC with charge loading 13.5 g/m and apex angle 50 ^{0} gives perfect severance of the target without any additional cracks. This paper describes the process methods, performance evaluation, test results and details of FLSC selection.

TE/O/06

Estimation of Drag Coefficient from Free Flight Data of an Artillery Rocket Using Neural Networks

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The proposed method to estimate drag coefficient from flight data of artillery rocket draws its inspiration from the research work done in the area of aerodynamic modeling and estimation of aerodynamic coefficients using Feed Forward Neural Networks (FFNNs). The universal mapping capability of FFNNs has been used to build flight dynamic model to predict acceleration for a given set of input variables of an artillery rocket in motion. The drag coefficient C_D was predicted by suitable interpolation and manipulation of the predicted acceleration for a given set of motion and atmospheric variables. The proposed method is first validated using simulated flight data of an artillery rocket. The final validation has been carried out using five sets of real flight data of a typical artillery rocket in motion. Further, the procedure using Maximum Likelihood (ML) method has also been applied on real flight data of these artillery rockets to estimate the values of drag coefficient at different Mach numbers. The estimated values of C_D obtained through the proposed method and the procedure based on ML method have been compared to evaluate the suitability of the proposed method. The results show that the proposed method can advantageously be applied on typical flight data of an artillery rocket in motion to estimate the drag coefficient C_D.

Calculation and Validation of Opening Shock of Slider-Reefed Parafoils

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The parafoil or ram-air parachute is a deformable that maintains its profile by trapping air between two rectangularly shaped membranes, sewn together at the trailing edge and sides, but open at the leading edge. After deployment when air starts filling it, the parafoil first goes though the canopy inflation process. Typically the instantaneous force applied during the canopy filling interval rises to a peak and then declines. The maximum opening force is known as the opening shock and represented by the dimensionless ratio, known as opening shock factor. It is the very important parameter to know the opening characteristics of any parachute and also in the selection of fabric material. The parameters which influence the opening shock factor are the mass ratio, canopy spreading rate, fall rate of the parachute-payload system at the time of line stretch and at the end of slider up phase. The calculation for opening shock factor is presented in the paper which is then validated from the data generated in technical trials using load-cell.

Spin Rate Extraction from a Single Lateral Accelerometer on a Space Probe

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The post flight analysis of impact/landing probe, including spin rate extraction, is an integral part of flight program management. Though, in launch vehicles, rate-gyros are used almost universally for gathering attitude information, for entry probe, trajectory sensors are chosen specifically to meet on-board space availability and power supply constraints while satisfying measurement accuracy requirements. In this context, it may be noted that magnetometers offer very attractive sensor mass and power consumption characteristics. Magnetometers are vector sensors, that is, they measure both the direction and magnitude of the external magnetic force field. They can operate over a wide range of temperatures and have no moving parts. From a single lateral magnetometer sensor it is possible to estimate spin rate as well as precession rate. But unfortunately, magnetometers are useless around celestial bodies which are without magnetic field (e.g. Moon, Mars, Titan etc.). Additionally, magnetic field around a magnetometer may be distorted by on-board electromagnetic interference.

Against this backdrop, it is found feasible to extract spin rate from accelerometer signal. Though accelerometers measure translational motion, its output is influenced by rotational motion. Consequently, accelerometers are suitably located on-board to excite the rotational component in their output signal. As a state of the art, accelerometers are being utilized to extract spin rate of different entry probes. It has been deduced that accelerometer should be placed sufficiently away from the centre of mass. A methodology, which combines the computation of indicative spin rate (from average acceleration and sensor location) with the estimation of power spectral density of accelerometer output signal, is developed. The proposed methodology is validated with simulated data and with the flight data of a typical sounding rocket.

Trajectory Reconstruction of An Inertialy Fixed Impact Probe to Moon

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The post flight analysis of impact/landing probe is an integral part of flight program management. For entry probe, trajectory sensors are chosen specifically to meet on-board space availability and power supply constraints while satisfying measurement accuracy requirements. In this context, accelerometers measure specific force, which is combined with a gravity model to obtain the spacecraft trajectory from de-boost point to surface impact through integration from a set of initial conditions (position and velocity). Accelerometers are successfully flown on Viking (to Mars), Pioneer, Galileo (to Jupiter), Pathfinder (to Mars), Huygens (to Titan) etc. While, accelerometer measurements are made in spacecraft-fixed frame, the integrations are easier to do in an inertial frame; and the results are best expressed in a rotating, planet-fixed frame. It is evident that attitude tracking is essential for converting the accelerations measured in the spacecraft frame into an inertial or planet-fixed frame. Gyroscopes measure angular rates, which are integrated to obtain the spacecraft attitude. But attitude measurement requires additional instruments onboard. In this connection, it is essential to explore various aspects of methodology for trajectory reconstruction (from only accelerometer signal), preferably with simulated data, before applying the same methodology to flight data.

Here, in order to test the applicability of a methodology, which utilizes the output from only two accelerometers (one each, along longitudinal and lateral body axes) a simulated synthetic mission dataset is prepared considering point mass-spherical moon gravity model from probe & orbiter separation point to probe impact point. The reconstructed trajectory is obtained by integrating only the acceleration profile. But for a very small de-boost duration, the trajectory is influenced only by moon gravity. This is evident from the absence of any sensed acceleration for most of the impact trajectory duration. By and large, the reconstructed trajectory matches quite closely with the simulated one. The differences in altitude and velocity are attributed to keeping the probe orientation fixed during de-boost operation, while in reality, owing to de-boost thrust misalignment, small change in probe orientation takes place during de-boost operation. The differences can be bridged with the availability of additional flight measurements (e.g. radar altimeter data).

Design Validation of Locking mechanism

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A heavy structure is required to be raised and firmly held in this position against dynamic and shock loads. The structure is also required to be retracted to its original position when need arises. The mechanism is to be remotely operated in sea environment and is to be accommodated in confined radial space.

A compact four bar linkage locking mechanism is designed to raise, hold and retract the heavy structure. The mechanism is remotely operated in marine environment using inbuilt hydraulic actuator and is expected to provide trouble free unattended service till next maintenance cycle. Holding the structure during shock in raised condition and retraction is a critical event and to ensure reliability a mechanical lock is provided. The total load of launcher is shared by three simultaneously operated mechanisms. The maintenance cycle is 3 years and shall be fully functional over this period.

Rigorous qualification testing is carried out to validate the design. Many difficulties were faced to match the required performance and are sorted by making minor changes.

This paper presents brief account of development of this critical mechanism including experimental work, difficulties faced and changes incorporated as outcome of test results to ensure performance parameters.

Studies on Ignition Mechanism of SFU Designs of Imported IR Flares and Indigenization

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The infra-red (IR) guided missiles pose a major threat to aerial weapon platforms. It is reported that during the last 20 years, 90% of aircraft losses were caused by IR guided missiles. Pyrotechnic IR decoy flares are used worldwide as a passive counter measure device to lure away the incoming IR guided missile. The IR decoy flare is an expendable item of counter measure dispensing system (CMDS). On receiving the threat command, the CMDS dispenses the appropriate number of flares in programmed manner. On ignition, IR radiation emitted by the flare being higher than the target aircraft, the missile homes on to flare instead of aircraft, thus provides maximum survivability in air to air or ground to air warfare. For building up the competence in the field of aerially delivered IR decoy flares, four different designs of imported (US, Israel and UK origin) flare were studied. Safety and Functioning Unit (SFU) is an important component of a flare that must prevent the ignition of a flare pellet in the flare container/dispenser loaded onto the aircraft. The explosive train of ignition mechanism of all the four flares has been found different. A systematic study of ignition mechanism of each flare/SFU design has been carried out and is presented in the paper. The interpreted merits and demerits of each design have been considered in indigenization of SFU for IR flare 218.

Parachute for 500 kg Class Weapon Systems

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Parachute system is integral part of modern airdrop able of any weapon. These system produce optimum performance for low terminal velocity at near vertical impact. These impact parameters can be achieved by using retarding tail unit and parachute. Parachute system provides deceleration, stability, and angle of attack so that optimum performance can be achieved. Depending upon the release speed release height one or two stage parachute is deployed in weapon system. Generally aircraft power source is used to activate the thermal battery of the weapon and its relese. After attaining the safe separation distance from the aircraft, the first stage parachute of the weapon is deployed with the help of pyrojack. After pre-set time the second stage parachute is deployed by separating first stage parachute. The laser based height sensor, then starts sensing the height and function at pre-set height from the target. The sensor provides all the electrical pulses for the operation of the weapon as per sequences. This paper discusses parachute characteristics, material selection and test plan which parachute has to go through during its development stage.

Design of Fixtures for Easy Integration and Dismantling of Low Caliber Submunition Type Rockets

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A few fixtures are designed to reduce the manpower requirement, improved safety of rocket integration and saving in integration time. There is necessity of less manpower for improved safety while working on explosives. Fully automatic techniques are available in the market but due to explosive safety considerations, it is very difficult to utilize them. The design of the fixtures presented in this paper fulfills above requirements. This paper brings out the design and working of various fixtures with mechanisms used for submunition warhead assembly, dismantling of filled rocket, integration of filled rocket, centre of gravity measuring fixtures and finally straightness measuring system for integrated rocket. All these fixtures use different mechanisms to achieve the desired functional requirement. Using these fixtures and mechanisms more than 400 rockets are integrated without any safety problems very successfully. The clamping mechanism, screwing & unscrewing mechanism, tilting, lifting mechanisms and straightness measuring system are discussed in this paper highlighting the main features.

Design and Development of Sighter Round for Submunition Warhead Rockets: Necessity and Applications

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To achieve high level of confidence to ensure that rocket with submunition warhead will function at proper range and desired height with pre determined time set on Electronic Time fuze in the absence of RADAR, this sighter round is designed. By using a modified explosive train accuracy and consistency for different fuse systems can also be assessed. By firing this sighter round the line of fire can be assessed and range table correction can be applied before firing actual round. There is inherent dispersion in free flight rockets. The trajectory from rocket to rocket can vary from round to round. Over and above, the meteorological variations may not get corrected in the corrections provided to the launcher in the absence of RADAR. So, before firing costly submunitions warhead in the PFFR ranges or even during warfare, it is very essential to ensure that rockets fall in the aimed target area. To achieve this, affordable sighter rocket system are very effective. Details of sighter system, design of its explosive train with smoke composition etc. are presented in this paper. The sighter round simulate mass, centre of gravity and mass moment of inertia properties of actual warhead. Static and dynamic trials conducted to evaluate the effectiveness of sighter round are also presented.

Design and development of a pump for the Xenon gas purifier for space application

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ASTROSAT mission has principal objective of multi wavelength studies of different classes of celestial sources and associated phenomena. Large Area X-ray Proportional Counter (LAXPC) is one of the major instruments on ASTROSAT suited for timing and low resolution spectral measurements of continuum X-ray emission from X-ray binaries and other classes of sources. The X-ray detection volume of LAXPCs will be filled with Xenon gas at a pressure of 2 to 3 atmosphere, is susceptible to electronegative impurities like oxygen, water vapour etc. The gas gain and energy resolution of the detector degrades when the impurity level increases to about 20 ppm or higher. Therefore a periodic purification of Xenon gas is required after launch of satellite in orbit to remove these impurities. Xenon gas is pumped through Oxisorb catridge, Xenon gas purifier system.

A pump is required to maintain a forced circulation of Xenon gas in the Oxisorb LAXPC circuit. This pump has to handle Xenon gas at the rate of 80 litres an hour at 2200 to 2500 torr pressure. The challenges involved were

1. Ultra low gas leakage
2. Absolutely no contamination from the moving surfaces and lubricants of the pump
3. More than 25 million cycles of maintenance free operation
4. Low power, mass and compact size
5. Pump performance should be almost immune to ambient pressure

To meet the above requirements, a configuration and design of the system has been worked out. The system involves a bellow type pump, driven by a Brushless DC motor and Drive Electronics.

After considering various aspects of environmental conditions and performance requirements, a twin bellow configuration has been chosen. The bellow pump is fixed in a 'U' bracket and linear actuation of the bellow is derived by means of an eccentric. The performance of the pump is independent of external pressure variation that eliminates fluctuation of load on the drive motor unlike in single bellow configuration.

This paper discusses in detail the configuration, design and performance of the above development. The tests performed on this pumping system with results are presented.

Development and Qualification of Flexible Seal based Thrust vectoring mechanism for Rocket Motor Nozzle

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Development and qualification of flexible seal based thrust vectoring mechanism for nozzle was completed. The system has been successfully tested and in use. This paper describes difficulties faced during the development and qualification process and tests carried out to qualify the system, interesting facts and experiences.

The flexible seal based thrust vectoring mechanism consist flexible seal and actuation mechanisms for thrust vectoring. Flexible seal is made of alternate layer of rubber and metal shims to provide required flexibility and to maintained structural integrity. Rubber materials were developed and qualified for the use. Critical dimensional inspection for concentricity, parallelism and angular mismatch was carried out. Pull test flexible seal to access the bond strength between metallic shims-rubber and to identify de-bond locations was carried out. Proof pressure test to confirm any leakage, deformation of the seal and distortion in shape was carried out. The flexible seal was subjected to vectoring test under pressure to measure the shift of thrust vectoring center, to measure spring torque and hysteresis characteristics. Integrated test of the system was carried out to evaluate the control system performance, to access the structural integrity of interface and to access the interference if any, between fixed and movable components during operation and to locate axis of rotation. Measurement of strains was carried out on shims and components subjected to loads to estimate margin of safety. Cyclic actuation test was carried out to estimate damping characteristics of the flexible system.

The system was tested with rocket motor firing. Temperature measurements were carried out to access the thermal loads at all critical locations of the system. Quick method to qualify the system was also established to reduce the qualification time. Method to achieve good bond quality between metal and rubber is established.

Smart Fuze: Solution for the Futuristic Munition

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A conventional fuze keeps the weapon safe and engages the target at predetermined time and place, leads to limited effectiveness and accuracy of the weapon. The basic principle of war was based on Protection, Mobility and Firepower. But present war cannot be thought of without Information and RAM-D along with the above parameters. The information and RAM-D calls for precision, accuracy and consistency in firing, which have become order of the day. To engage omni-directional threats in the warfield of reducing response time with minimum collateral damage requires a munitions with built-in flexibility to effectively engage and prioritise variety of targets. This provokes the thought process for incorporation of smartness in munition. When you say that the munition is 'smart' it is all about the fuze. A 'smart fuze' can be defined as a fuze, which is accurate, reliable, with target identification capability, multi functionality and self-health monitoring feature. A smart fuze possessed with logic to engage the most important target, technology to function in desired mode (i.e. point detonating super quick, point detonating delay, time, proximity and course correction), technology of miniaturisation to reduce the size of mechanical sub-systems, power source and the best packaging technology of the electronic gadgets to sustain environmental as well as launch forces. To realise the smart fuze the major technologies are MEMS based sub systems, Miniaturise power source, Copper explosive in chip form, Embedded smart sensors and Encapsulation to sustain weapon environments. This paper will describe the details of the various advance technologies to realise the smart fuze.

Design & Analysis of Vertical Handling System for Rocket

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Delicate & heavy Rocket components are required to be maneuvered in vertical position during various stages of rocket or missile development. Rocket is required to be lifted for assembly & disassembly during manufacture stages. It also requires installation & retraction from launch pad in vertical position. Rockets consist of delicate electronic components, which should not get damaged or malfunctioning. Vertical Handling system is required for tasks integration of the rocket with its launch platform. The Rocket need to be accurately position at over location to have smooth engagement with umbilical connection to power the rocket. Handling system is provided with capabilities of universal joint to maintain fixed angular rotation. A dedicated handling system is used for performing these tasks for the particular rocket.

The paper describes one such handling system designed by the author for handling of rockets. The Rockets are lifted using handling system, with the help of a crane. Handling system consists of lower & upper handling assembly. Combination of disc springs has been incorporated in handling device to take care of damage to the electronic components due to vibration & accidental fall. Final mating of the umbilical connection is solved with help of a precision lowering actuator. Handling assemblies are qualified for various load conditions and the complete system is proof tested to validate the design. The handling system has been successfully used in handling rockets.

Design and development of a pump for the Xenon gas purifier for space application

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TE/P/16

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TE/P/17

Smart Fuze: Solution for the Futuristic Munition

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TE/P/19

Static and Dynamic Sealing Techniques for Pyrodevices for Launch Vehicle Applications

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Pyrodevices utilize the energy contained in explosive charge to carry out the desired mechanical function. They are used in launch vehicles and satellites to perform various mission critical operations. These devices contain different interfaces, which should be properly sealed for its effective functioning. The seal requirement can be static or dynamic in nature. Due to the extreme pressure and temperature existing in pyrodevices during its actuation, design of these seals are very difficult and there are no strict guidelines for the same. Elastomeric seals are generally used for this purpose for devices working at or near ambient temperature. However for pyrodevices designed for low temperature applications, elastomer based seals cannot be used as they undergo glass transition. For such devices to have effective sealing, a completely different design methodology is to be adapted. Special features like application of various types of metallic bellows, membranes, gaskets, interference fit and juidicious provision of

Welding and brazing have to be provided at suitable locations. At the same time, the number of joints have to be kept minimum for reliability and processability. This paper explains the basic methodology of the sealing selection for pyrodevices and the influence of operating conditions. It also deals in detail, the various design features and evaluation procedures employed in low temperature application pyrosystems, where elastomeric based seals are not permitted.

Development of Tactile Sensor for Space Debris Impact Detection in Manned Missions

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The outer space is filled with numerous unwanted objects, including meteors, meteorites and parts of old satellites. They are collectively called Space Debris. They exist in various size ranges and are posing increasing collision risk to man-made satellites, International Space Station and manned missions. Various protection strategies have been developed for various size ranges of these objects considering a lot of factors like mission profile, debris size, density and distribution in the operating zone. These include Whipple shields and collision avoidance maneuvers. For debris sized > 10cms orbital tracking is available and hence avoidance is possible. For debris sized < 1 cm the protective shield is effective. For debris sized between1cm - 10cms the damage potential is large and hence efforts are being made to develop effective protection methodologies. Debris in this size range on collision with space craft can perforate the pressurized walls. This can lead to mission criticality and catastrophe in manned exploration flights, often jeopardizing the astronaut safety. Lack of efficient techniques for prompt detection of the debris impact and triangulation of the position of the perforation holes, demands the crew to abandon the punctured module. Therefore, it is important to install an active detection system in addition to passive protection systems like Whipple Shields, for in time location of perforation damages and to aid in quick repair. In this paper, we propose a highly flexible and modular yet simple and practical perforation hole detection system for space debris impact. This embedded system uses a conductive grid film attached to the pressurized wall. The sensing intelligence and algorithm are stored in a microcontroller. The proposed design has the added flexibility of being able to differentiate between no perforation, partial perforation and complete perforation. It can also provide impact rate, impact area estimate, impact rate estimate, damage estimate etc. The effectiveness of this detection system was verified by numerical analysis and hypervelocity impact simulation. A concept demonstration model of size 15x15cm has been implemented and tested successfully. Further to this, a feasibility study for implementing this concept in a generic manned mission module was carried out. Finally, we have proposed a generic design methodology to aid in future research in this area.

Pressure Monitoring System for Solid Motors

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In most of the Launch Vehicles and Missiles Solid Propulsion has an important role to play. In Launch Vehicles Solid Stages are used to provide the initial thrust lift-off as the core thrusting stage and / or as booster stages. In some of the Launch Vehicles solid stages are used as high performance upper stages. A well designed solid stage is expected to follow the nominal pressure-time curve for the stipulated mission duration within the nominal bounds. The performance solid motors are mainly assessed by the pressure-time curve during their functioning. During the development and qualification phase of the solid motors pressure monitoring is required in addition to thrust measurement during static firing test. Also during proof pressure stesting of motor cases pressure monitoring has to be carried out.

Pressure monitoring is a very critical system in solid motors. The scheme used for pressure monitoring has to be very reliable. The system has to ensure leak tightness during the various phases of solid motor preparation as well as in flight as any failure will be catastrophic and will lead to mission failure. The scheme shall ensure ease of assembly and should be operator independent. This paper details the evolution of such a reliable scheme based on shaft O-rings for pressure monitoring of solid motors. As the new scheme has to be implemented in operational launch vehicles it had to undergo elaborate qualification program. The various testing carried out to qualify the new scheme are discussed in detail.

TE/P/22

Parametric Cycle and Engine Performance Analysis of Aircraft Engines

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Cycle analysis studies the thermodynamic changes of the working fluid as it flows through the engine. It is divided into two types of analysis - parametric cycle analysis (on-design analysis) and engine performance analysis (off-design analysis). Parametric cycle analysis determines the performance of engines at different flight conditions and values of design choice and design limit parameters. Engine performance analysis determines the performance of a specific engine at all flight conditions and throttle settings. In both analyses the components of an engine are characterized by the change in properties they produce. In an ideal engine, working fluid behaved as a perfect gas with constant thermal properties whereas for the real engine analysis the thermal properties are considered as a function of temperature.

In the present work, the on- design analysis of Ideal and real aircraft engines, such as, Ramjet engine, Turbojet engine with/ without afterburner, Turbofan engine with convergent nozzle (separate exhaust), Mixed flow Turbofan engine and Turboprop engine are carried out. The off-design analysis of real aircraft engines, such as, Turbojet engine with/ without afterburner, Turbofan engine with convergent nozzle (separate exhaust) and Turboprop engine are also carried out and compared with the on-design analysis.

This study is to develop a general versatile computer code in Fortran language which can be used to carryout the on and off design analysis of the above aircraft engines. Each subroutine will make the analysis of each engine as per the requirements and data supplied. Based on the mathematical modeling a versatile computer code has been developed in Fortran language for the parametric cycle and engine performance analysis of major types of real aircraft engines mentioned above. All subroutines are merged to form a single general purpose code so that it can be used for analysis of above mentioned real engines.

A comparison of thermal, propulsive and overall efficiencies are made among the various aircraft engines at various Mach numbers, compressor pressure ratio and work output coefficients. The efficiencies of real engines are compared with ideal engine. In the case of turboprop engine the variation of work output coefficient with compressor pressure ratio is made. The comparison of performance of different real engines is also carried out. The developed code can be used to compare the performance of currently available aircraft engines coming under the above seven categories. As a case study the parametric cycle analysis of a typical military aircraft engine (TF34-GE-100) is carried out and the results are compared with ideal engine.

TE/P/23

Micro Arc Oxidation (MAO) coating for Gun Barrel applications.

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Small arms are subjected to extensive firings and exposed to extreme climatic conditions prevailing in different terrains. The weapons need to be highly reliable in the field environment so as to instill high degree of confidence in the minds of the soldiers. Majority of weapon components are designed by selecting high strength steels to ensure high reliability of operation. Barrel is one of the most critical components in any form of Small Arms, which projects the projectile towards target. It is also a heavy component & accounts for about one third of the total weight of the weapon. An effort is made to make the barrel of a grenade launcher in Aluminium alloy to make the weapon lightweight as well as maintenance free. A suitable surface protection was required to be provided for inside surface of the barrel to enhance its life as well as ease of maintenance. It was essential to identify suitable surface engineering methodology.

Aluminum alloys are hard anodized for various applications, which has its own limitations like lower hardness and integrity with base metal. Hard anodizing is therefore not suitable for inside bore applications since it is subjected to very high wear and tear and also to very high flame temperatures of the order of 2000 K. A new technology of coating named 'Micro Arc Oxidation (MAO)' has been explored for coating of inside barrel bore for the first time for Armaments applications in India.

MAO is basically a chemical conversion process in which α-phase Aluminium oxide is diffused into substrate. It gives a variable increasing hardness from inner to outer diameter of barrel bore. The hardness achieved is more than 1200VPN. The coating has an edge over the conventional hard anodizing which has got relatively softer γ-phase Aluminium oxide.

The MAO coated Gun was subjected to repeated/rigorous firing of 50 grenades and it was observed that the MAO coating has withstood the firing without any kind of adverse effects. The barrels have been studied using magnifying microscope at critical sections from metallurgy point of view. The results of the same in comparison to hard anodic coating barrels have been presented in this paper.

With this, a new era of surface engineering has been established for Small Arms, which can be extended to various other Armament applications too.

TE/P/24

Development of Compressor and Turbine Blades of Helicopter Engines Through Rapid Prototyping and Virtual Manufacturing Technique

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Spare management has always posed numerous challenges in overhauling of aero engines. This becomes more complex in collaborated projects especially in the absence of manufacturing technology. Overhauling of Helicopter engines is no exception. Considerable rejection of compressor and turbine blades due to erosion, corrosion & burning in a Helicopter engine is an impediment in maintaining the fleet. Total dependence on OEM for the supply of critical spares always results in costs and time overruns. Hence, indigenous development of critical spares like, compressor and turbine blades becomes essential especially when these spares are no more in the manufacturing range of OEM.

This paper deals with the development and certification approach for compressor and turbine blades of a typical turbo shaft engine. It consists of two parts, namely part I & part-II. While part-I deals with the development methodology, second part deals with the airworthiness requirements for certification with few case studies on development of compressor and turbine blades.

Part-I: Development of 3D models, 2D drawings & patterns was carried out using laser scanning and Rapid Prototyping Technology (RPT). A suitable methodology was devised to validate the process for development of component drawings. Simulation of castings was done using ProCAST software and forging simulation was carried out using DEFORM software.

Part-II deals with the airworthiness requirements that include Theoretical evaluation, Specimen level testing, Process evaluation, Component level testing and Engine testing. The theoretical evaluation involves stress and vibration analysis using ANSYS software, development of Goodman Diagram to evaluate the fatigue life, development of Campbell diagram for frequency analysis. Specimen level evaluation involves characterization for its chemical composition and mechanical properties. Process evaluation includes chemical composition, macro & microstructure, mechanical properties and NDE to assess its structural integrity. Fatigue test following Armstrong method has been adopted to assess the fatigue life of blades and for comparison against the original cat-A blades. An AMT (Accelerated Mission Test) has been formulated considering various flight profiles the engine is subjected to during exploitation in its TBO/TTL. Finally the finished blades assembled in an engine are subjected to AMT for evaluation and certification.

TE/P/25

Performance Evaluation of Space Based Instrument Systems Utilizing Aerial Survey Aircraft as a Test bed

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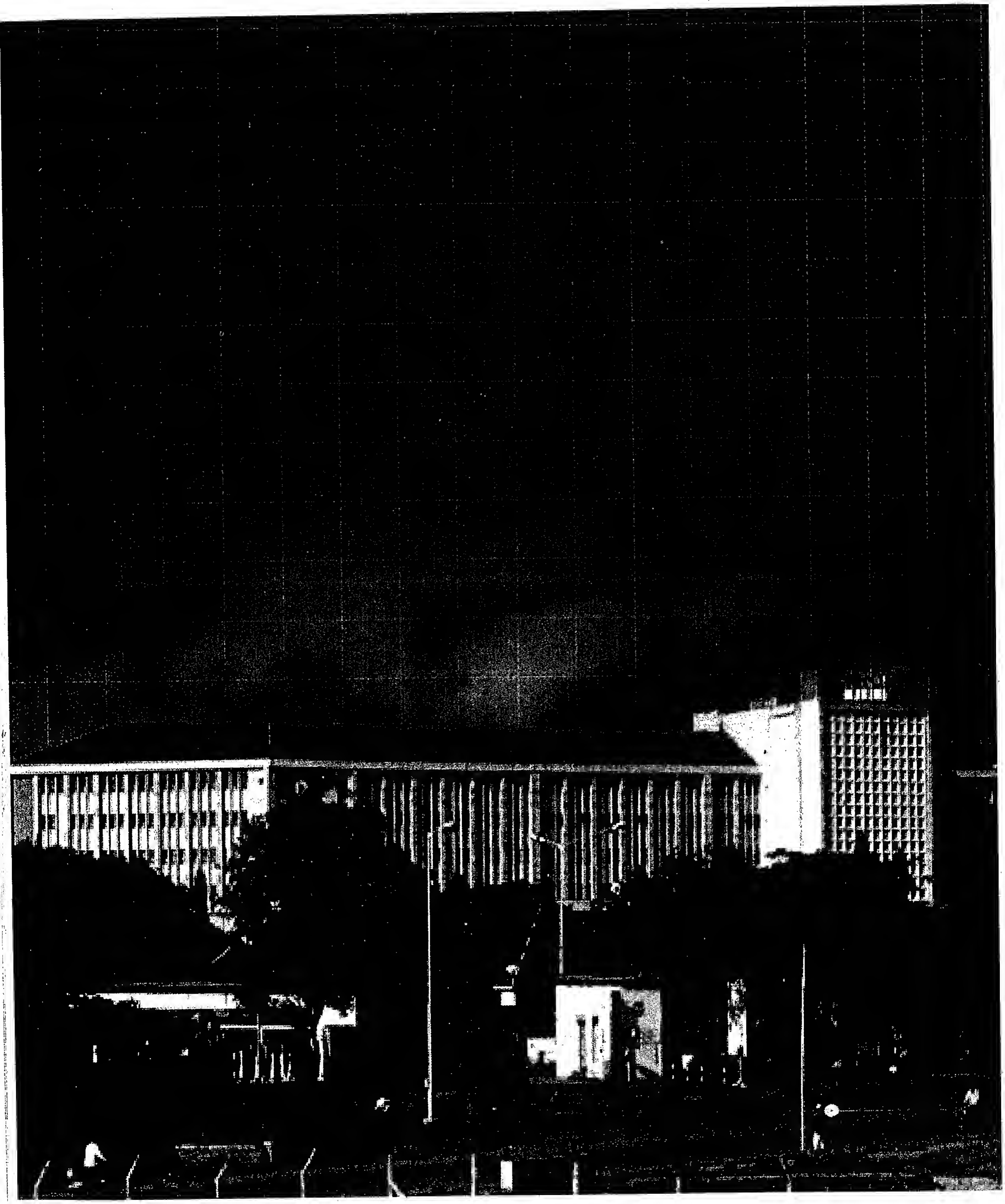
A wide range of instruments are required to be developed for navigation, attitude determination and altitude ranging for spacecraft applications. After completion of ground testing in laboratory, these instruments need to be tested in conditions similar to the actual flights in which they are required to operate.

Aerial survey aircraft belonging to the National Remote Sensing Agency (NRSA) has been successfully utilized as a test bed to checkout INS-GPS system developed by ISRO Inertial Systems Unit (IISU). Flight path trajectory data from independent sources such as aircraft INS and KGPS systems in real time or after post processing was available for comparison of results with INS-GPS system under development. The other advantages of aerial survey aircraft are (i) facility for the designer to observe the instrument performance in actual flight conditions, (ii) facilities comprising of cutouts in the belly of aircraft, data acquisition systems, built in power supply modules, vibration damping mounts, instrument racks etc. for installation of systems (iii) communications systems for interface with ground systems and (iv) low lead time for preparatory aircraft modification and task planning. Weight and power consumption do not pose any challenges as space systems are designed to be light and efficient and survey aircraft can comfortably carry a heavy payload.

Limitations of this test procedure include restricted flight velocity and attitude acceleration rates when compared with space craft and altitude ceiling. However, checkout of interface elements between instrument sub systems, data acquisition and processing with on board computers can be carried out very effectively and economically using survey aircraft as a test bed.

TE/P/26



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